AD NUMBER

AD901651

NEW LIMITATION CHANGE

TO

Approved for public release, distribution unlimited

FROM

Distribution authorized to U.S. Gov't. agencies only; Test and Evaluation; FEB 1972. Other requests shall be referred to Air Force Aeronautical Systems Division, Attn: SDQH, Wright-Patterson AFB, OH 45433.

AUTHORITY

asd, usaf ltr, 17 apr 1974

FTC-TR-7

AD INEXPLORATA

AFFTC

VOLUME I OF II

CATEGORY II PERFORMANCE TEST OF THE UH-1N HELICOPTER

ROBERT H. SPRINGER
Project Engineer

DONALD BERGER
Lieutenant Colonel, USAF
Project Pilot

TECHNICAL REPORT No. 72-17

MAY 1972

See form 1473

Distribution limited to U.S.Government agencies only (Test and Evaluation), February 1972. Other requests for this document must be referred to ASD (SDQH), Wright-Patterson AFB, Ohio 45433.

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE

DEPARTMENT OF THE AIR FORCE

HEADQUARTERS AERONAUTICAL SYSTEMS DIVISION (AFSC)

WRIGHT-PATTERSON AIR FORCE BASE. OHIO 45433

REPLY TO ATTN OF: ASD/SDQH 6-42 (Maj Thompson/54480/bjs/R&D 13-2-3N)

SUBJECT: ASD Addendum to FTC-TR-72-17, UH-1N Performance Tests

TO: Recipients of FTC-TR-72-17
(Initial distribution on attached list)

This report is a part of and should remain attached to FTC-TR-72-17, "Category II Performance Test of the UH-1N Helicopter". As noted in Volume I of the report, Volume II contains only detailed technical data which substantiates the test results presented in Volume I. Therefore, Volume II has been printed and distributed only to selected addressees who have demonstrated engineering interest in the UH-1N system. The paragraph numbers below correspond to the recommendations in the AFFTC Technical Report, Volume I.

1 through 7. Concur with intent. ASD has initiated action to incorporate the required information in the appropriate aircraft manuals.

8. Concur with intent. ASD has forwarded this recommendation to the Airframe Contractor for review and comments.

FOR THE COMMANDER

William D. Eastman Jr

WILLIAM D. EASTMAN, Jr, Lt Col, USAF
Chief, Helicopter Program Office
Directorate of Combat Systems
Deputy for Systems

1 Atch
a/s

H-1 TEST REPORT DISTRIBUTION

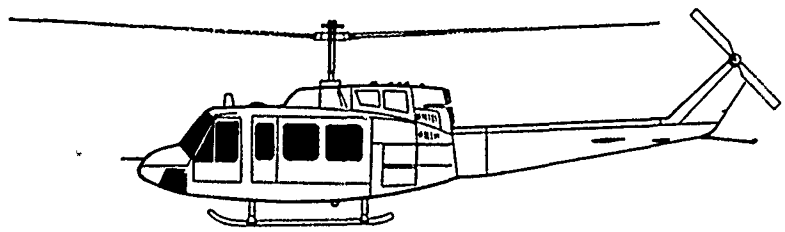
HQ USAF: RDQ/RDPN/PRPL/SAMI/SMEMA/SSSRA
HQ AFSC: DOV/SDNS/DLTA
HQ ASD: SDQH*/SMPL/XRHD
HQ AFLC: MMEA/MMPC
AFFTC: 6510 TGE*
WRAMA: MMD-1*/MMEAP*
OCAMA: MMEAA
USAFSOF: DR
1SOW/DM, 317 SOS, 834 FMS (EGLIN AFB AUX #9, FLA)
HQ MAC: DOQ
HQ FTD: ENDI/ENTE
AFFDL: FGC
ADTC: DLOSL/DLGZ/DLYE
SAAMA: MMLO
HQ TAC: DRLS/DMML/DOSO
HQ ARRS: ARXRD
USA AVSCOM: AMSAV-R-F
USA AVSYS TEST ACTIVITY: SAVTE-P (EDWARDS AFB, CALIF)
USA WECOM: AMSWE-REW
USA AE&SW: LIBRARIAN
DCASO (OTTOWA 6, ONTARIO, CANADA)
1 CANADIAN LOG LIAISON: MCLRCA*
BELL HELICOPTER COMPANY*
UNITED AIRCRAFT OF CANADA, LTD*
FAA: MS-110/FS-160*
DEFENSE DOCUMENTATION CENTER*
USDA FOREST SERVICE
THE RAND CORP: LIBRARY D
SANDIA CORPORATION LIBRARY
1550ATTWG: DO
HQ ATC: TTME/TTAT-B
STTC: TSOP-E/TSOP-T/37ST104
HQ COMD: DOT/DMM, 1 COMP WG: WDO/DOHL
HQ SAC: OAI
USAFA: DFAN
AU: AUL/LSE-6389
1002 IG GP: IGDSFR
USAF SAM: SMKEN
CINCPACAF: DORQ
HQ USAFE: DOL
HQ 17AF: DOOW, 7TH SOS: SOSO
USAFSO: OOT/MME
NASC: AIR-503-D, -5104C2, -5362/OAP-31*
NATC: CT-14/FT 2312*
NADC: ADL
USA BPA: SAVBE-E

(*Denotes recipients of substantiating data reports)

FTC-TR-72-17

VOLUME I OF II

CATEGORY II PERFORMANCE TEST OF THE UH-1N HELICOPTER



ROBERT H. SPRINGER
Project Engineer

DONALD BERGER
Lieutenant Colonel, USAF
Project Pilot

Distribution limited to U.S. Government agencies only (Test and Evaluation), February 1972. Other requests for this document must be referred to ASD (SDQH), Wright-Patterson AFB, Ohio 45433.

FOREWORD

Testing was conducted between 17 November 1970 and 16 February 1972 at Edwards AFB, Bakersfield, Bishop and the nearby high altitude test site of Coyote Flats, California, and in Canada at Canadian Forces Base Cold Lake, Alberta. The tests were conducted under authority of AFFTC Project Directive 69-49B (Program Structure 443N). UH-1N helicopter USAF S/N 68-10776 was utilized for these tests.

The authors of this report wish to express their appreciation to Sergeant W.T. Geary, Jr., Technical Sergeant J.C. Dixon, Miss Nancy Hart, Mr. P.W. Martin, and Captain R.J. Taylor, Canadian Armed Forces, for their assistance in data reduction and engineering analysis. In addition, the dedicated efforts of Mr. D. Abramowitz, crew chief, and his maintenance crew are gratefully acknowledged. Special thanks and appreciation are expressed to Mr. John Somsel, project manager, for his guidance and assistance.

Foreign announcement and dissemination by the Defense Documentation Center are not authorized because of technology restrictions of the U.S. Export Control Acts as implemented by AFR 400-10.

Prepared by:

Robert H. Springer

ROBERT H. SPRINGER
Project Engineer

Donald Berger

DONALD BERGER
Lieutenant Colonel, USAF
Project Pilot

Reviewed and Approved by:

9 MAY 1972

James W Wood

JAMES W. WOOD
Colonel, USAF
Commander, 6510 Test Wing

Robert M. White

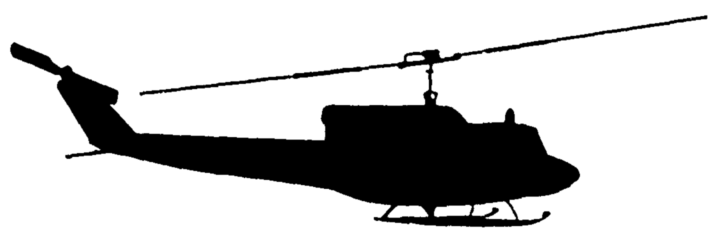
ROBERT M. WHITE
Brigadier General, USAF
Commander

ABSTRACT

This report presents the results of the UH-1N Category II performance tests conducted to obtain data for the Flight Manual. In general, hover, climb, level flight, and takeoff performance equalled or exceeded that estimated in the Flight Manual; the exception being level flight at low weight and/or altitude. Level flight tests with external armament resulted in a 5- to 10-percent reduction in range capability depending on loading. The UH-1N had excellent single-engine performance resulting in a relatively small AVOID area on the height-velocity curve. A single-engine go-around was possible at all conditions outside a well defined CAUTION area. Slope landing tests were made on slopes up to 17 degrees. The standard airspeed system would not register airspeeds below 15 to 20 knots, and there were position errors of up to 9 knots in level flight and 7 knots in climb. Discrepancies in the engine power indicating systems were found to be sufficient to possibly cause an unnecessary replacement of a satisfactory engine.

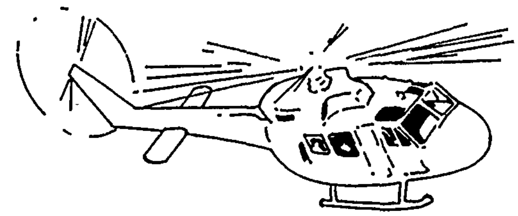
table of contents

*Appendixes I and II will be published in volume II.



list of illustrations

Appendix I, Volume II



Appendix II, Volume II

list of tables

list of abbreviations and symbols

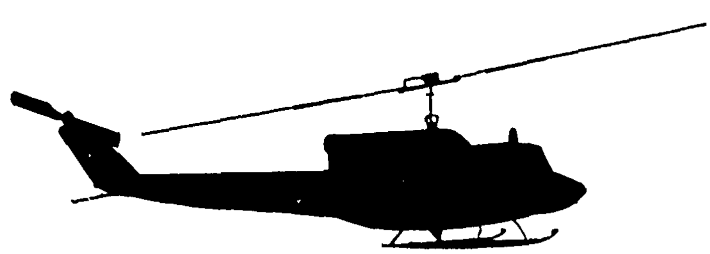
Item	Definition	Units
A	rotor disk area	ft^2
BHC	Bell Helicopter Company	- - -
C	centigrade or Celsius	- - -
C_P	power coefficient	dimensionless
C_T	thrust coefficient	dimensionless
CIT	indicated compressor inlet total temperature (t_{t_2})	deg C
cps	cycles per second	- - -
FAT	free air temperature (t_a, ambient air temperature)	deg C
GW	gross weight	lb
H_p	geopotential altitude (pressure altitude)	ft
ΔH_{pc}	correction for static source (altimeter) position error	ft
h	tapeline altitude	ft
ICAO	International Civil Aeronautics Organization	- - -

Item	Definition	Units
IGE	in ground effect	- - -
ITT	inter turbine temperature	deg C
K	Kelvin	- - -
KCAS	knots calibrated airspeed	- - -
KIAS	knots indicated airspeed (corrected for instrument error)	- - -
KTAS	knots true airspeed	- - -
m	local slope of a curve	variable
M_{TIP}	advancing blade tip Mach number	dimensionless
NAMPP	nautical air miles per pound of fuel	- - -
NAMT	nautical air miles traveled	- - -
N_E	engine output shaft speed	rpm
N_f	power turbine speed (N_2)	rpm
N_g	gas producer speed (N_1)	rpm
N_R	main rotor speed	rpm
OAT	outside air temperature ($t_{t_{ic}}$ indicated total temperature corrected for instrument error)	deg C
OGE	out of ground effect	ft
PA	pressure altitude	ft
P_a	atmospheric or ambient pressure	in. Hg
P_3	compressor bleed air pressure for fuel control	psi
P_{t_2}	compressor inlet total pressure	in. Hg
Q	engine output torque	ft-lb
R	rotor radius	ft
R_e	Reynolds number	dimensionless
R/C	rate of climb	ft per min
R/D	rate of descent	ft per min
RN_E	referred engine output shaft speed $(\frac{N_E}{\sqrt{\theta_a}})$	rpm
S/E	single-engine operation	- - -
shp	shaft horsepower	$\frac{550 \text{ ft-lb}}{\text{sec}}$
SL	sea level	- - -
T	temperature (always used with a subscript)	deg K
T_a	ambient or atmospheric temperature	deg K
UACL	United Aircraft of Canada, Limited	- - -
V	velocity (used in general terms)	kt
V_c	calibrated airspeed	kt

Item	Definition	Units
V_i	indicated airspeed (corrected for instrument error)	kt
V_{NE}	indicated airspeed never to exceed	kt
V_t	true airspeed	kt
ΔV_{pc}	correction for airspeed position error	kt
W	gross weight	lb
W_f	fuel flow	lb per hr
δ_a	ambient pressure ratio $(= \frac{P_a}{P_{a_{SL}}})$	dimensionless
δ_{t_2}	engine compressor inlet pressure ratio $(= P_{t_2}/P_{a_{SL}})$	dimensionless
Δ	incremental change in a parameter	variable
ψ	rotor blade azimuth (zero at the tail, 180 deg at the nose)	deg
θ_a	ambient temperature ratio $(= T_a/T_{a_{SL}})$	dimensionless
θ_{t_2}	engine compressor inlet temperature ratio $(= T_{t_2}/T_{a_{SL}})$	dimensionless
μ	rotor advance ratio	dimensionless
Ω	rotor angular velocity	rad per sec

Subscripts

Item	Definition
a	ambient
i	indicated
s	standard day conditions
t	test day conditions or total
SL	sea level
2	engine station (compressor inlet face)
3	engine station (compressor)
5	engine station (inter turbine)



INTRODUCTION

The UH-1N is a twin-engined, single-main-rotor helicopter which is basically a military version of the civilian Bell Helicopter Model 212. The primary missions for which the UH-1N was procured are the Tactical Air Command's Special Operations Forces missions of counter-insurgency, unconventional warfare, and psychological operations. The secondary missions are the transport of personnel and equipment and the delivery of protective fire by the installation of appropriate weapons. The UH-1N armament system consists of pintle-mounted 7.62mm miniguns (XM-93), 40mm grenade launchers (XM-94), and rocket launchers (LAU-59/A) with seven 2.75-inch folding fin rockets per pod.

The Category II performance test program was conducted by Air Force Flight Test Center (AFFTC) personnel. The first flight was on 17 November 1970, and the last flight was on 16 February 1972. The performance test data were acquired utilizing UH-1N USAF S/N 68-10776. Two hundred fifty-one sorties for 277 flight hours were required for 162.8 test hours. The extra hours were required for ferry, functional check flights and other flying in direct support of the test program. The test aircraft was also used for flying qualities and propulsion testing concurrently with the performance testing and accumulated a total of 335 sorties and 432.8 flying hours during the Category II test effort.

The test aircraft sustained major damage in an accident on 16 February 1972 during height-velocity testing at Edwards Air Force Base. The Category II performance testing terminated on that date. As a result, the planned single-engine takeoff tests at 2,300 feet PA, autorotational descent tests above 10,000 feet, and height-velocity tests at 9,500 pounds and 10,500 pounds at 2,100 feet PA were not completed.

The UH-1N helicopter has a single two-bladed main lifting rotor and a tractor tail rotor instead of the more conventional pusher tail rotor. The UH-1N utilizes the basic UH-1D fuselage and is equipped with thin tip main and tail rotor blades. The aircraft is powered by a United Aircraft of Canada Limited T400-CP-400 power package consisting of two PT6T-4 free-turbine turboshaft engines coupled to a combining gearbox having a single output shaft. Each engine has an uninstalled rating of 900 shaft horsepower at sea level, standard day conditions. Overrunning clutches in the combining gearbox allow engine torque to be transmitted in one direction only, thus providing for both single-engine operation and two engine-out autorotation. Load sharing between the two engines is equalized by an automatic torque matching device. The maximum allowable forward speed is 130 knots indicated airspeed (KIAS), and the maximum gross weight is 10,000 pounds. The maximum allowable gross weight for testing purposes was 10,500 pounds and 11,500 pounds for tethered hover tests. Except for a short period, the maximum allowable altitude was 15,000 feet. The empty weight of the test helicopter, including test instrumentation, was 6,733 pounds. The production UH-1N has an empty basic gross weight of approximately 6,000 pounds.

The original installed calibrated test power package was gearbox S/N 4064 with engines S/N 66127 and S/N 66128. This power package was replaced on 5 August 1971 when a power degradation was observed. A new calibrated test power package, gearbox S/N 4061, with engines S/N 66121 and S/N 66122, was installed. On 11 September 1971, engine S/N 66121 was replaced when a broken inter-turbine-temperature (ITT) lead fitting was found. The new left engine was S/N 66126.

TEST AND EVALUATION

Pitot-Static System Calibration

The pitot-static system calibration tests were conducted to determine the position error of the standard and test (boom) airspeed and altimeter systems. Both airspeed systems were calibrated in level flight by the tower fly-by and ground-speed course methods. The standard system was calibrated in climb and autorotational descent by comparing it to the boom system. The standard system static source position error (altimeter) was determined by the tower fly-by and ground-speed course methods in level flight only. The calibrations were conducted in the clean configuration only. Airspeed calibration summary curves are shown in figures 1 through 3, and the calibration test data are presented in figures 1 through 3, appendix I.

The standard airspeed system position error in level flight was within specification limits (MIL-I-5072A, reference 1) at speeds above 39 knots indicated airspeed. Zero error existed at 67 KIAS and the error was 2 knots or less from 51 to 125 KIAS. This speed range covers the level flight maximum endurance (loiter) and recommended cruise (maximum range) speeds. The error at airspeeds less than 39 KIAS was greater than specification limits (±4 knots). At 30 KIAS the error increased to 6 knots. Airspeed did not indicate below 15 knots. Flight in the speed range of 20 to 40 knots is normally limited to takeoff and landing, but many mission requirements necessitate relatively slow flight, frequently close to the ground.

The standard airspeed system position error was in excess of the specification limits below 70 KIAS in climb and below 75 KIAS in autorotation. Best climb speed was 48-52 knots indicated airspeed (KIAS) which required a position error correction of 6 knots. Minimum rate of descent autorotation speed is 57 KIAS which requires a position error correction of 9 knots. The correction at the airspeed for maximum range in autorotation was 4 knots or less.

The large airspeed position error in low speed level flight, climb, and autorotational descent was attributed to the location of the standard system pitot-static source. The pitot-static source head was located on the upper forward fuselage over the cockpit area. This location was subject to heavy rotor wash influence at low speeds in forward flight and in climbs. In autorotation the pitot-static source is subject to severe airflow disturbance due to the large angle between the relative airflow and the fuselage. Means of reducing airspeed indicating errors should be investigated. **(R 8)**[1]

The static source position error resulting in an altimeter error was within specification limits at indicated airspeeds greater than 45 knots. Below this speed the altimeter error was greater than the specification limits. However, the maximum error was less than 25 feet, and is acceptable.

---

[1]Boldface numerals preceded by an R correspond to the recommendation numbers tabulated in the Conclusions and Recommendations section of this report.

UH-1N USAF S/N 68-10776
T400-CP-400 Engine
Category II

STANDARD SYSTEM

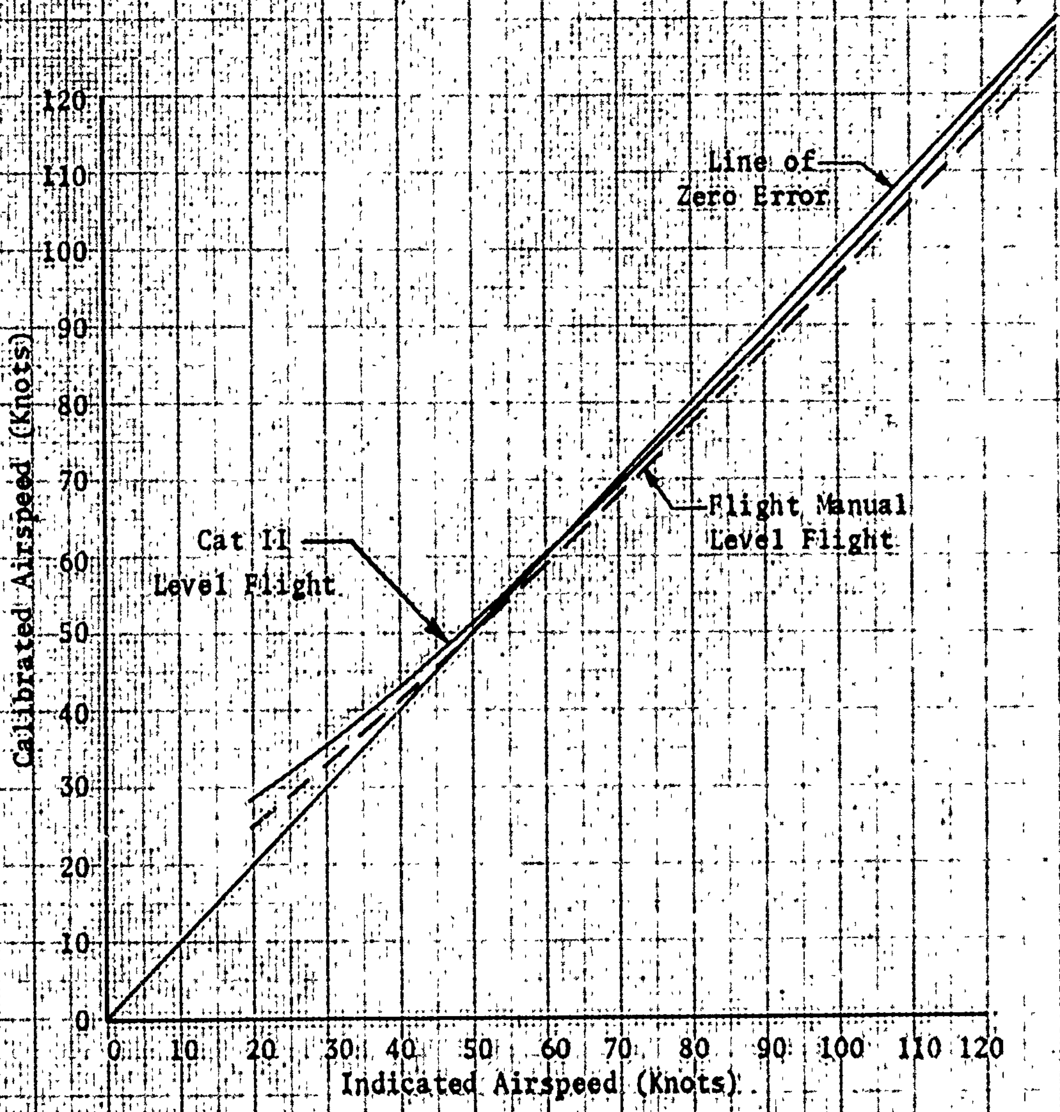


Figure 1 Airspeed Calibration Summary-LEVEL FLIGHT

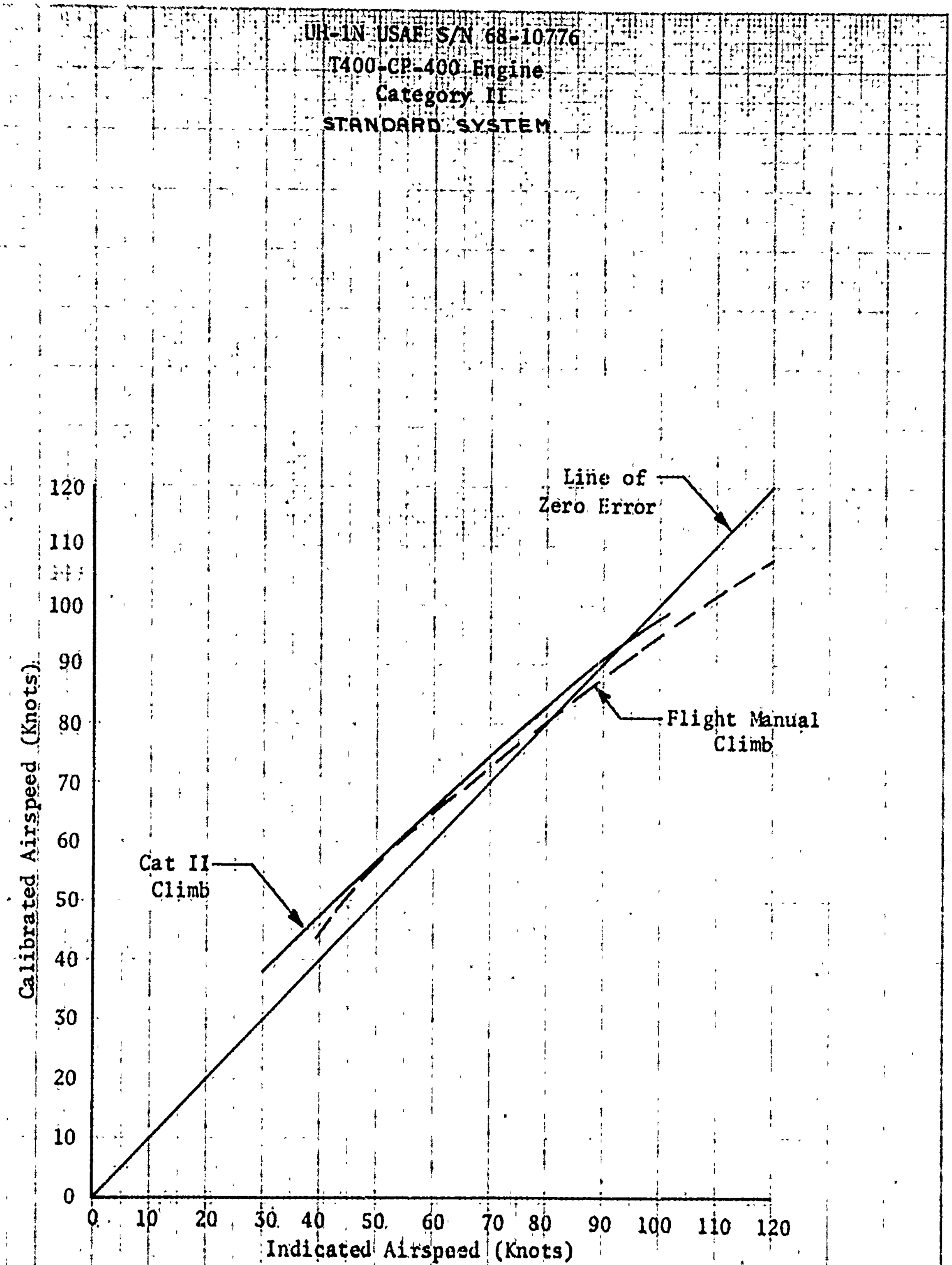


Figure 2. Airspeed Calibration Summary - CLIMB

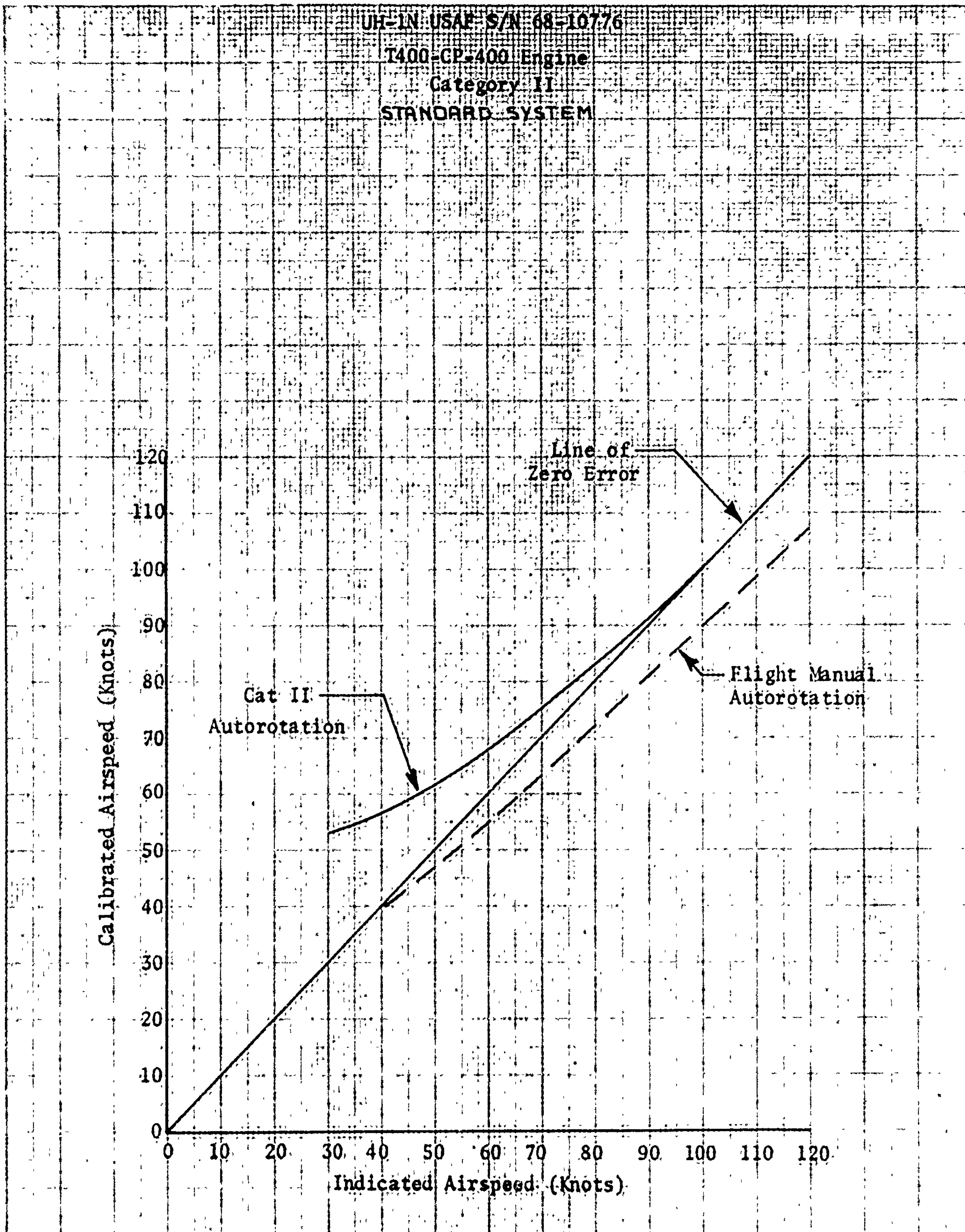


Figure 3 Airspeed Calibration Summary - AUTOROTATION

Hover Performance

In-ground-effect (IGE) and out-of-ground-effect (OGE) hovering performance data were obtained by tethered and free flight techniques at average pressure altitudes of 2,000 and 9,600 feet. Tethered hovering was investigated at skid heights from 2 to 60 feet in less than 3 knots of wind. Free flight hovering was done at all tethered skid heights plus 100 feet skid height.

Special attention was given to determining increases in power required due to rotor blade compressibility. The increase in power required due to compressibility was undetectable at low gross weight, altitude, and skid heights. The compressibility effects became noticeable as the gross weight, altitude and/or skid height were increased. For the typical mission condition shown in figure 4, the power required increased approximately 9 percent as the temperature varied from 40 degrees C to 0 degrees C. The estimated power required figures in the Flight Manual (reference 2) were 7 to 8.5 percent higher than test data for corresponding blade tip Mach numbers. The nondimensional hovering performance data are presented in figures 4 through 13, appendix I.

Hovering ceiling performance was better than estimated in the Flight Manual. Table I compares test and Flight Manual hovering performance. A free flight hover OGE ceiling test was made at the following conditions: 10,090 pounds gross weight, -6.4 degrees C ambient temperature, 320.5 rpm rotor speed. The OGE hover ceiling was 8,710 feet PA. The Flight Manual gives a ceiling of only 4,500 feet for these same conditions. Calculated hover ceiling summaries for standard day and hot day conditions are shown in figures 5 and 6. Table II shows computed single-engine hovering performance.

It was determined that the UH-1N OGE boundary varied from skid heights of 55 to 57 feet as the thrust coefficient varied from minimum to maximum, respectively (figures 4 through 7, appendix I). In order to ascertain that 60 feet skid height was sufficient to be OGE, free flight hover data were obtained at 100 feet skid height.

The tethered hovering tests were conducted with constant referred rotor speeds ($N_R/\sqrt{\theta}$), and because of the temperature ranges experienced during the tests, main rotor speeds were varied from 324 rpm down to as low as 294 rpm. Increasingly greater tail rotor thrust was required as the main rotor speed was reduced. This resulted from more main rotor shaft torque being required to maintain a constant shaft horsepower as the main rotor shaft rpm was reduced, and more thrust was required of the tail rotor to counter the main rotor shaft torque. Directional pedal positions as a function of main rotor thrust coefficient (C_T) are presented in figures 14 through 17, appendix I. Left directional pedal travel was never lower than 14.5-percent (1 inch) remaining for the tethered hover tests at main rotor speeds of 314 rpm (94 percent) to 324 rpm (100 percent). Main rotor speeds less than 314 rpm resulted in more left pedal travel required, and at high C_T and low rpm combinations, 10 percent or less of pedal travel remained. The good directional control characteristic is attributed to the large chord tail rotor blade and tractor tail rotor installation.

Table I

HOVERING CEILING TEST AND FLIGHT MANUAL COMPARISON

Standard Day

Military Power[1]

Rotor Speed 324 rpm (100 percent)

Pressure Altitude (ft)	4-Ft Skid Height		15-Ft Skid Height		OGE	
	Flight Manual Gross Weight (lb)	Test[2] Gross Weight (lb)	Flight Manual Gross Weight (lb)	Test[2] Gross Weight (lb)	Flight Manual Gross Weight (lb)	Test[2] Gross Weight (lb)
SL	*	*	*	*	*	*
5,000	*	*	*	*	9,950	10,430
10,000	*	*	10,150	10,200	9,250	9,550

*Gross weight limited to 10,500 pound maximum.

[1]Derived from figure 163, appendix I.

[2]Test gross weight calculated for conditions stated.

Table II

SINGLE-ENGINE HOVERING PERFORMANCE[1]

Standard Day

Military Power

N_R 314 rpm (97 pct)

Altitude	Maximum Gross Weight (lb)	
	4-Foot	OGE
SL	9,190	7,820
5,000	8,200	7,000

[1]Computed values: power required derived from figure 13, appendix I; power available derived from figure 163, appendix I.

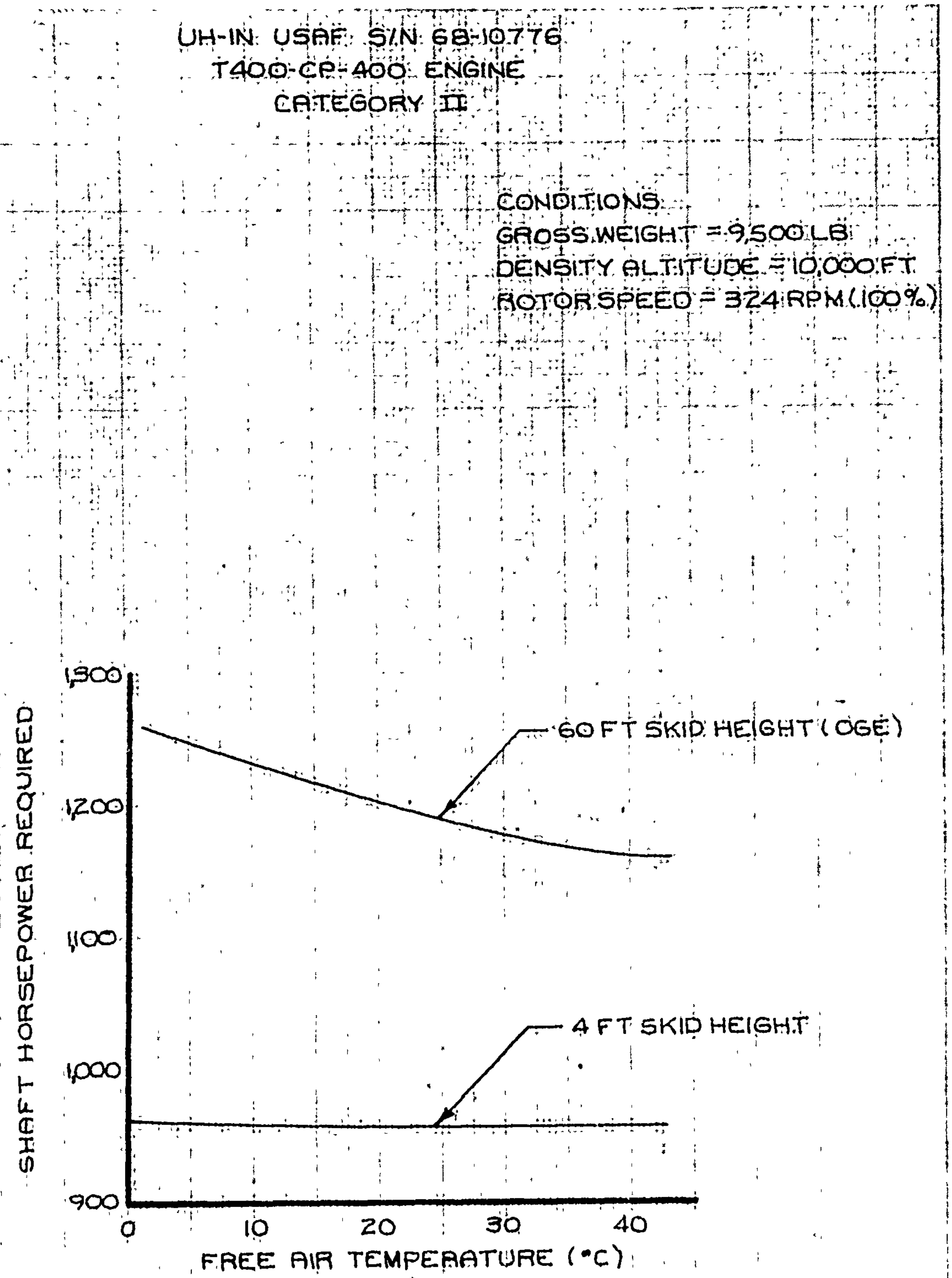


FIGURE 4. COMPRESSIBILITY EFFECTS ON POWER REQUIRED TO HOVER

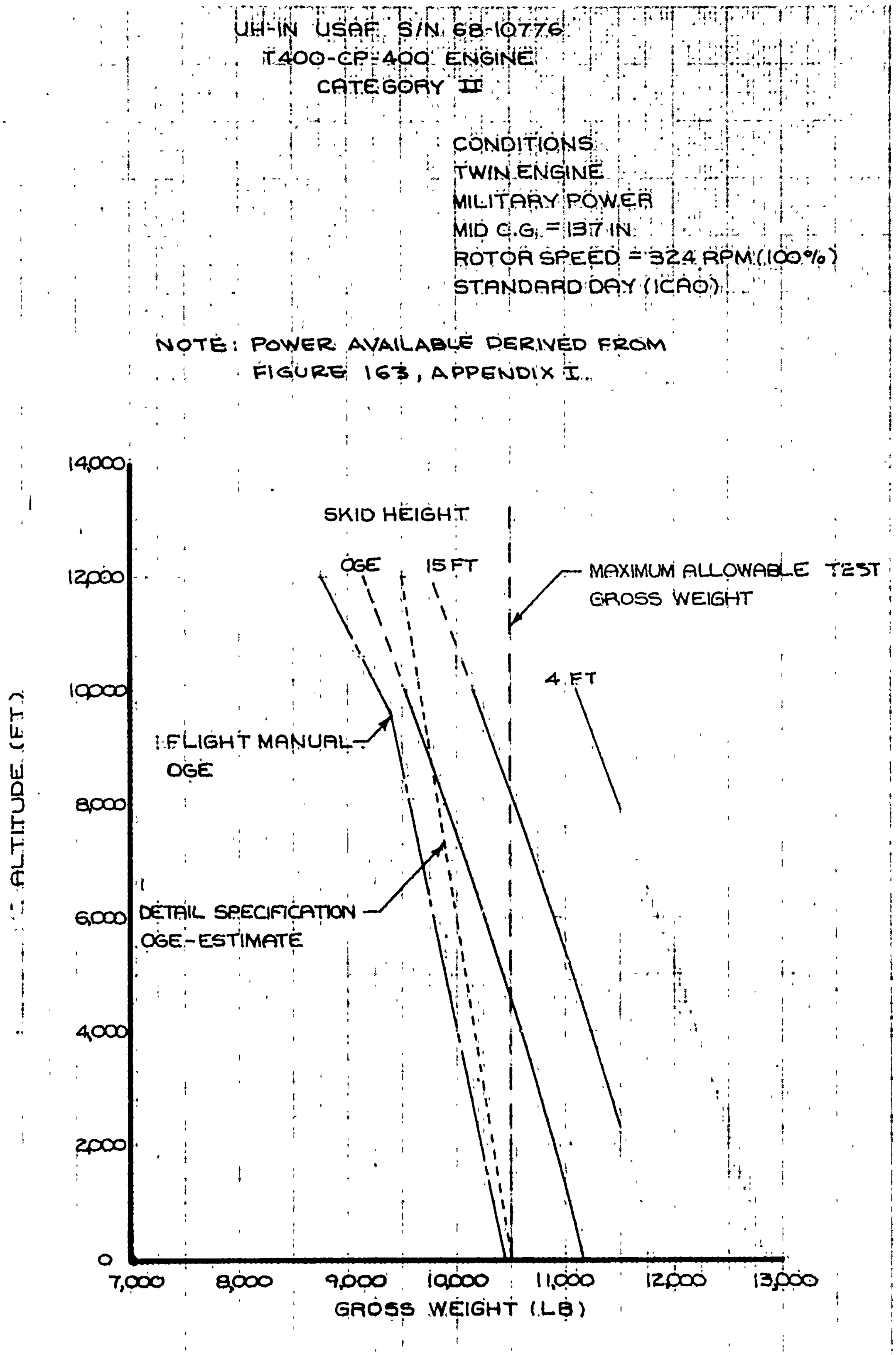


FIGURE 5. HOVER CEILING SUMMARY - STANDARD DAY

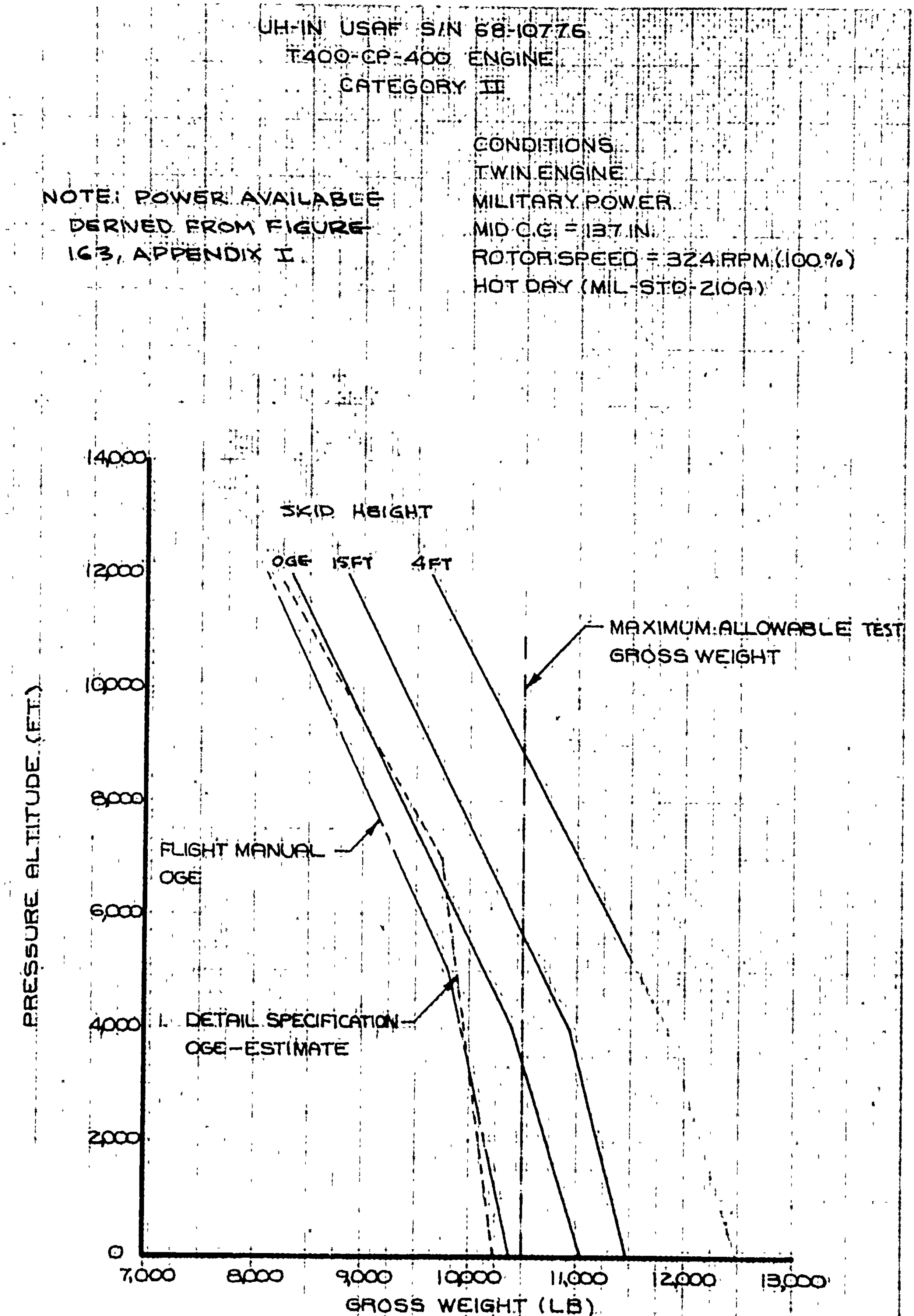


FIGURE 6. HOVER CEILING SUMMARY - HOT DAY

Takeoff Performance

Takeoff tests were performed at a pressure altitude of approximately 9,600 feet (10,500 feet average density altitude). The test gross weight was varied from 9,000 pounds to 10,500 pounds. The rotor speed was 324 rpm (100 percent) at the start of all takeoff tests. Five basic takeoff techniques were used:

1. Level acceleration from a 4-foot hover without rotor rpm bleed.
2. Level acceleration from a 4-foot hover with rotor rpm bleed.
3. Climb and acceleration from light-on-skids without rotor rpm bleed.
4. Climb and acceleration from light-on-skids with rotor rpm bleed.
5. Level acceleration from a 15-foot hover without rotor rpm bleed.

These techniques are described in detail in appendix I.

Figure 7 compares the takeoff performance for level acceleration from a 4-foot hover and climb and acceleration from light-on-skids techniques for one set of conditions. The climb and acceleration takeoff technique always yielded shorter distances to clear a 50-foot obstacle than the level acceleration. The typical reduction in distance was 100 feet. The use of rotor bleed with either method always resulted in a reduction in distance required. The maximum performance was attained using rotor rpm bleed with the climb and accelerate technique. This resulted in a 150-foot (30 percent at 25 KIAS) reduction in distance compared to level acceleration with no bleed. Figures 18 through 37, appendix I, present the takeoff test data.

As can be seen, with the conditions in figure 7, a vertical takeoff and climbout were possible when using the rotor rpm bleed technique, but this was not possible when rotor rpm bleed was not used. The advantage of the N_R bleed technique over maintaining 100 percent N_R was that it assured topping power was always reached. Present topping procedures are based on 97 percent N_R. Maintaining 100-percent N_R did not always assure that engine limits of N_g and ITT were reached. Since these limits are independent of one another, the pilot must be aware of all of them and assure that these limits are not exceeded.

The vertical climb takeoff poses one major hazard; considerable flight exposure in the avoid area of the height-velocity curve. Therefore, the vertical climb takeoff should not be used except when operational necessity requires this and the risk involved in operating in the height-velocity avoid area is acceptable.

The standard airspeed system was unreliable below 20 KIAS. Consistent minimum takeoff distances in this speed range could only be accomplished by holding a pitch attitude.

The level acceleration from a 15-foot hover technique simulated a takeoff with a sling load. The technique involved was basically the same as the level acceleration from a 4-foot hover. The performance for a given set of conditions was less, however, since more power was required to hover at 15 feet than at 4 feet. Therefore, less excess power was available for the takeoff acceleration and climbout.

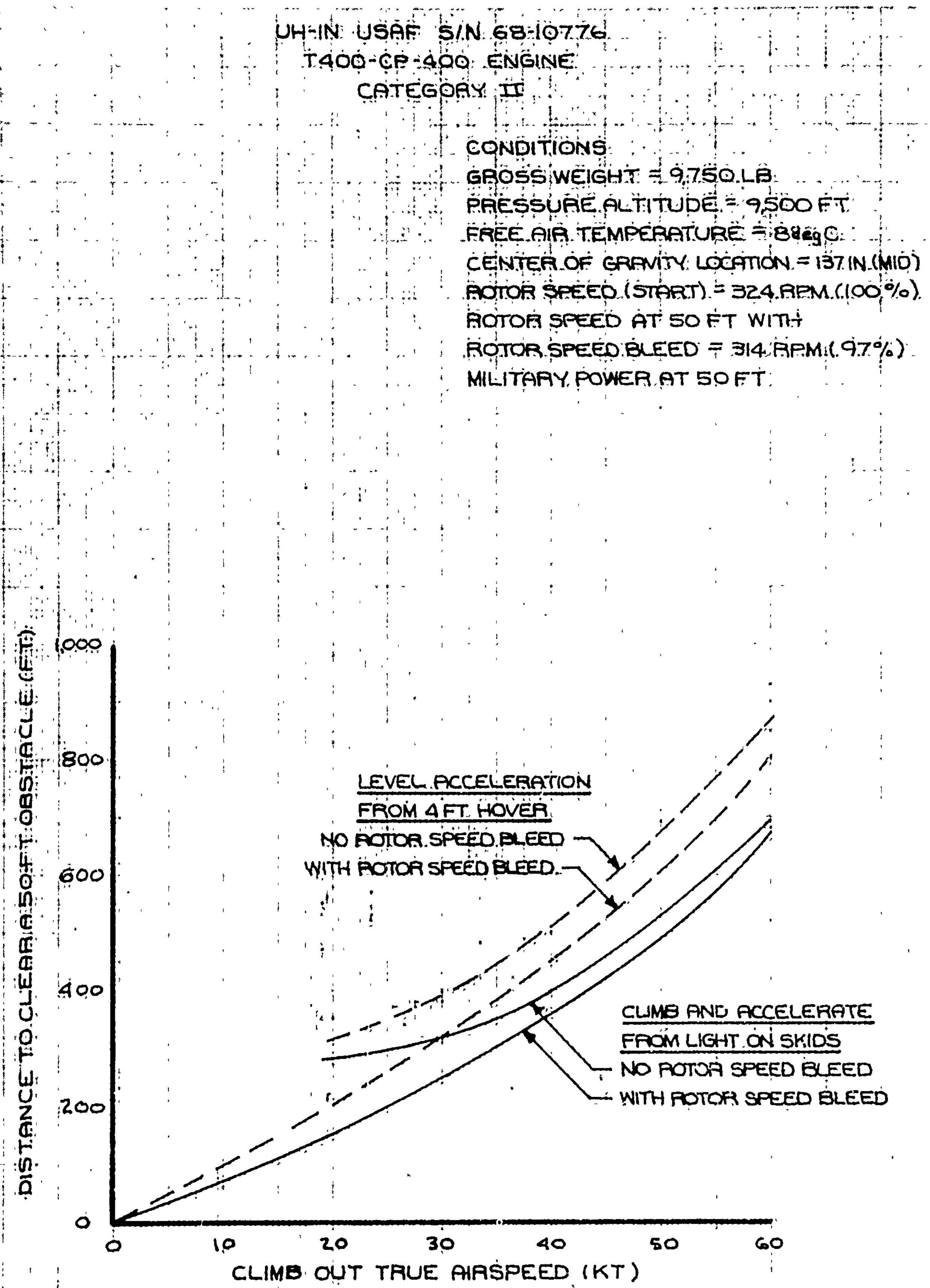


FIGURE 7. TAKEOFF PERFORMANCE SUMMARY

Section II of the Flight Manual should be changed to incorporate the following: (R 2)

TYPES OF TAKE-OFF (Change to Read)

The factors governing the type of take-off to be accomplished are the gross weight of the helicopter, the pressure altitude, outside air temperature, wind direction and velocity, the size of the take-off area, and the tactical situation. The most commonly used types of take-off are take-off to a hover, normal take-off and maximum performance take-off. Maximum performance take-off techniques are level acceleration, climb and acceleration and the slide take-off. For all techniques, coordinate the collective pitch and N_f governor (beep) switch as required to maintain the desired rotor speed. The N_R may overshoot when the beep switch is held continuously at full increase. Overshoot of N_R can be controlled by "pulsing" the switch as collective pitch is increased until maximum N_R is reached. N_R is then easily controlled with collective pitch. Rotor rpm bleed may be used with any of the three maximum performance techniques. The advantage of using rotor rpm bleed over maintaining 100-percent N_R is that it assures topping power is always reached. Take-off distances listed in Appendix I are for maximum performance take-offs with and without rotor bleed techniques when maximum allowable (or available) power is maintained until clear of obstacles and climbout airspeed is maintained in accordance with Appendix I.

TAKE-OFF TO A HOVER (Remains unchanged)

NORMAL TAKE-OFF (Remains unchanged)

MAXIMUM PERFORMANCE TAKE-OFFS (Change to read)

Level Acceleration

The level acceleration take-off may be required when operating from small and/or restricted areas when sufficient power to hover out of ground effect is not available. From a hover, accelerate as rapidly as possible to the climb-out airspeed, reference Appendix I, maintaining maximum power at 100% N_R. Climb out at this airspeed, maintaining maximum power until clear of all obstacles, then smoothly increase airspeed to normal climb speed.

Rotor "Bleed"

If a shorter distance to clear an obstacle is required, use the level acceleration technique combined with rotor bleed. As the climb airspeed is reached and the climb started, slowly bleed the rotor rpm from 100% to 97%. As the rotor rpm is bled, care must be taken not to exceed engine limitations. As the obstacles are cleared, lower the

collective pitch slightly to allow N_R to increase to 100% while maintaining airspeed. After 100% N_R is obtained, establish normal climb.

WARNING

During climb-out do not let the airspeed drop below translational lift speed (approximately 15 knots) or aircraft settling may occur. If the aircraft starts to settle, lower the nose of the aircraft slightly to increase airspeed.

Climb and Acceleration

From a light-on-the-skids condition, smoothly increase power to maximum while maintaining N_R at 100%. At the same time use cyclic stick to obtain the desired climb-out airspeed in accordance with Appendix I.

NOTE

Due to the unreliability of the airspeed indicator below 20 KIAS, use pitch attitude change to establish the take-off.

Approximately 5 degrees of nosedown pitch change will give 25 KIAS climb-out speed.

After the desired height has been reached, smoothly transition to a normal climb. Vertical takeoffs and climbs to out-of-ground effect are not recommended; however, if operational necessity requires a vertical climb, accelerate to minimum climb airspeed as soon as possible after clearing obstacles.

Rotor "Bleed"

If the absolute maximum performance is required for take-off, the rotor bleed technique may be used in conjunction with the climb and acceleration. As the aircraft passes through a 10 foot height, slowly bleed the rotor from 100% to 97%, taking care not to exceed engine limits. When sufficient altitude for obstacle clearance is obtained, lower collective slightly to regain 100% N_R and then transition to normal climb speed.

WARNING

During climb-out do not let the airspeed drop below translational lift speed (approximately 15 knots) or aircraft settling may occur. If the aircraft starts to settle, lower the nose of the aircraft slightly to increase airspeed.

Slide

Apply maximum allowable power and move the cyclic control stick forward far enough to obtain take-off airspeed. The helicopter will normally fly itself off the surface at translational lift speed. When airborne, use cyclic stick to correct any nosedown pitching while accelerating to the climb-out airspeed.

WARNING

During climb-out do not let the airspeed drop below translational lift speed (approximately 15 knots) or aircraft settling may occur. If the aircraft starts to settle, lower the nose of the aircraft slightly to increase airspeed.

CROSS WIND TAKE-OFF (Remains unchanged)

Climb Performance

Twin-engine climb performance to 15,000 feet PA was determined for gross weights of 8,500 and 10,000 pounds at maximum continuous power (88-percent torque) and 314-rpm (97-percent) rotor speed. The UH-1N helicopter was limited to a pressure altitude of 15,000 feet at the time these tests were conducted because the flight envelope had not been approved above that altitude. The test data are presented in figures 38 through 41, appendix I.

The installed engines were essentially "flat-rated" up to 15,000 feet at the temperatures encountered during the climb tests. A torque of 88 percent (maximum continuous power) could be maintained throughout this altitude range.

The calibrated airspeeds for maximum rates of climb ranged from 53 to 59 knots. An average climb airspeed of 56 KCAS (50 KIAS) was satisfactory for all weight and altitude combinations. The characteristics of the sawtooth climb data indicated that since the curvature of the fairing in the area of the maximum rate of climb was relatively flat, a

deviation in climb speed of 3 KCAS from best climb speed will not significantly reduce the rate of climb. The establishment of a single climb speed for all conditions will make the pilot's task easier. The Flight Manual recommends 55 KIAS (60 KCAS) based on estimated performance. Holding an exact airspeed during the continuous climbs was tedious due to the relatively neutral static longitudinal speed stability. Any slight upsetting inputs allowed the airspeed to change slightly with very little tendency to return to the trimmed airspeed. However, since the speed changes encountered were generally less than ±3 knots and no difficulties ensued, only a slightly increased pilot workload was required to maintain the desired airspeed.

Climb tests with external armament were not flown. However, the minimum-power-required airspeeds shown in the speed-power performance tests indicated that these airspeeds ranged from 52 to 65 KCAS with two LAU-59/A rocket launchers only, and from 50 to 63 KCAS with full external armament. Based on the climb characteristics, an average climb speed of 56 KCAS is acceptable for all external armament conditions. Therefore, a single climb speed of 56 KCAS is recommended for all climb conditions, clean or with external armament.

Level Flight Performance

Level flight performance tests were conducted to determine power required and specific range (nautical air miles per pound of fuel, NAMPP) over a wide range of gross weights (W), pressure altitudes, and free air temperatures (FAT). The tests were conducted with controlled advancing blade tip Mach number so (at a constant gross weight, airspeed, and density altitude) the power-required increase due to compressibility effects could be determined. All basic test conditions were flown twin-engine with a clean mid cg loading. Single-engine conditions were flown to investigate the increased range and loiter time potential. Level flight performance tests were flown with external armament installed to determine the effects on power required, fuel flow, and specific range. Level flight performance tests were flown with full forward and aft cg locations to determine their effects on power required.

Figures 42 through 101, appendix I, present the level flight nondimensional data.

Compressibility

The level flight tests of the UH-1N showed that for a given gross weight, airspeed, rotor speed, and density altitude, the power required increased with increasing advancing blade tip Mach number above the critical Mach number. Figure 8 illustrates the increased power required for one set of conditions.

The increase in power required due to compressibility was also more pronounced at higher altitude and/or gross weight. This was due to an increased main rotor blade angle of attack, which resulted in a lower critical Mach number.

Tests indicated that for referred rotor speeds ($N_R/\sqrt{\theta}$) up to 310 rpm, no increase in power required due to compressibility occurred. Therefore, for all forward speeds and a rotor speed of 314 rpm (97 percent), free air temperatures down to 23 degrees C would not result in a

power-required increase due to rotor blade compressibility. For temperatures below 23 degrees C, power required could increase up to 26 percent.

Figures 9 through 12 illustrate the effects of temperature on level flight range and loiter performance.

Twin-Engine Operation

Maximum Range Performance

Tests showed that specific range performance at sea level compared very closely with that estimated in the Flight Manual. However, as the cruise altitude was increased, the test specific range (NAMPP) was much greater (improved) than that estimated in the Flight Manual. Table III illustrates the differences between test and Flight Manual.

Table III

COMPARISON OF TEST AND FLIGHT MANUAL LEVEL FLIGHT PERFORMANCE

Gross Weight = 9,000 pounds

Standard Day Conditions

N_R = 314 rpm (97 percent)

Mid cg

Pressure Altitude (ft)	Maximum Endurance[1] (Loiter)					Maximum Range				
	Fuel Flow (lb/hr)		V_c (kt)		Test Percent Increase in Fuel Flow	NAMPP[2]		V_t (kt)		Test Percent Increase in Specific Range
	Test	Flight Manual	Test	Flight Manual		Test	Flight Manual	Test	Flight Manual	
SL	490	480	67	65	-2.0	0.183	0.180	116	116	+1.5
5,000	460	420	65	62	-9.5	0.200	0.190	118	111	+5.5
10,000	410	415	57	58	+1.0	0.209	0.181	111	102	+15.7

[1]Maximum endurance calculated at minimum power required.

[2]NAMPP is at speeds for 0.99 maximum NAMPP or V_{NE}, whichever is less.

Figures 9 through 12 summarize the twin-engine level flight performance test results. Optimum cruise altitude for maximum range increased as gross weight was decreased. Figure 13 summarizes the optimum altitudes for maximum range cruise for various gross weights and temperature conditions.

Figure 14 summarizes the effects of density altitude and compressibility on maximum range cruise performance for various FAT's at a gross weight of 9,000 pounds.

Maximum Endurance (Loiter)

Test results agreed very closely with the Flight Manual estimates for fuel flows and calibrated airspeeds for maximum endurance at all gross weights and altitudes except at or near 5,000 feet. Table III and

figures 9 through 12 show the test result summaries and comparisons with the Flight Manual estimates. Compressibility effects at minimum-power-required speeds (maximum endurance) were apparent, but the difference in fuel flow was within 20 pounds per hour throughout the temperature range of cold to hot day as is seen in figures 9 through 12. Figure 15 shows the power-required variation due to compressibility for a typical maximum endurance flight condition for various FAT's.

Single-Engine Operation

Single-engine level flight performance tests were conducted at conditions duplicating twin-engine tests to compare maximum range and endurance. Power required under identical flight conditions was the same as for twin-engine operation. However, the single-engine fuel flow was lower than twin-engine fuel flow for the same total shaft horsepower. Table IV and figures 16 through 18 compare single- and twin-engine operation level flight performance for the maximum range and endurance conditions.

Since the engine power section output shafts operate through a common combining gearbox having a single drive shaft to the rotor transmission, no asymmetrical forces result from single-engine operation.

Maximum range may be increased more than 25 percent by single-engine operation for long range cruise. Loiter time may also be increased by an average of 25 percent by single-engine operation. When the operational situation requires the maximum possible range and/or loiter time, single-engine operation should be used. **(R 3)**

Table IV

SINGLE- AND TWIN-ENGINE OPERATION LEVEL FLIGHT PERFORMANCE COMPARISONS

Gross Weight = 9,500 pounds

Standard Day Conditions

N_R = 314 rpm (97 percent)

Mid cg

Pressure Altitude (ft)	Maximum Endurance Fuel Flow[1] (lb/hr)			Maximum Range					
				NAMPP[2]			True Airspeed (kt)		
	Twin Engine	Single Engine	Difference (pct)	Twin Engine	Single Engine	Difference (pct)	Twin Engine	Single Engine	Difference (pct)
SL	495	350	-29.1	0.155	0.206	+36.6	100	90	-10.5
5,000	450	340	-24.5	0.185	0.233	+25.9	112	104	-7.6
10,000	425	335	-21.2	0.196	0.253	+29.1	109	106	-3.3

[1]Maximum endurance calculated at minimum power required.

[2]NAMPP is at speed for 0.99 maximum NAMPP or V_{NE}, whichever is less.

External Armament

Level flight performance tests were flown with external armament installed to determine the power required and specific range data. The tests were flown with two different loadings:

1. Rocket pods only - two LAU-59/A rocket launchers, mid cg, and cargo doors closed.

2. Full armament - two LAU-59/A rocket launchers, two XM-93 miniguns extended fixed to fire forward, mid cg, and cargo doors open.

A comparative summary of the results of the armament tests for one condition is shown in table V.

Figures 19 through 21 present a summary of the armament tests compared with the clean loading. Specific range reduction for the rocket-pod-only loading remained fairly constant up to 10,000 feet PA, averaging 5 percent. Specific range reduction with full external armament averaged 10 percent up to 10,000 feet PA. Table VI shows an example of the variation in power required for the clean, rocket-pods-only, and full armament loadings.

Maximum possible range with full armament installed may be achieved by flying enroute to an operational area with the XM-93 miniguns stowed inside and the cargo doors closed. This could result in a 3 to 4 percent greater specific range potential than with the guns extended.

Table V

LEVEL FLIGHT PERFORMANCE WITH EXTERNAL ARMAMENT

Gross Weight = 10,000 pounds

Hot Day (MIL-STD-210A)

N_R = 314 rpm (97 percent)

Mid cg

Pressure Altitude (ft)	Maximum Endurance (Loiter)[1] Fuel Flow (lb/hr)					Maximum Range[2] (NAMPP)				
	Clean	Pods Only	Increase (pct)	Full Arm	Increase (pct)	Clean	Pods Only	Decrease (pct)	Full Arm	Decrease (pct)
SL	525	538	2.5	545	3.8	0.181	0.174	-3.3	0.165	-8.8
5,000	470	497	5.7	500	6.4	0.188	0.180	-4.3	0.172	-8.6

[1]Maximum endurance calculated at minimum power required.

[2]NAMPP is at speed for 0.99 maximum NAMPP or V_{NE}, whichever is less.

Table VI

LEVEL FLIGHT POWER REQUIRED INCREASES WITH EXTERNAL ARMAMENT

Gross Weight 9,500 pounds

V_T = 100 Knots

N_R = 314 rpm (97 pct)

MIL-STD-210A Hot Day

Altitude (ft)	Clean SHP_{req}	Rocket Pods Only SHP_{req}	Percent Increase Over Clean Condition	Full Armament SHP_{req}	Percent Increase Over Clean Condition
SL	598	673	6.5	710	18.6
5,000	595	644	7.6	671	12.7
10,000	702	782	11.4	838	19.3

Center of Gravity Location

Level flight performance tests were flown to determine the effects of cg location on power required. The aft cg location (sta 142.9) resulted in small power-required differences when compared with the mid cg (sta 137) data. A forward cg location (sta 130.1) resulted in significant power-required increases.

The power-required increase with a forward cg location resulted in a calculated 5-percent reduction in specific range. The aft cg condition resulted in an insignificant change in specific range.

Vibration

Vibration data were obtained during level flight performance tests. The flights selected for the vibration analysis provided two flight envelope cross sections: (1) Maintaining a constant coefficient of thrust (C_T) with the referred rotor speed ($N_R/\sqrt{\theta_a}$) varying from 300 to 340 rpm to evaluate compressibility effects, and (2) maintaining a constant $N_R/\sqrt{\theta_a}$ with varying C_T's from 0.0032 to 0.0050 to evaluate gross weight and altitude effects. The forward and aft center of gravity location level flights were evaluated to determine the effect of these conditions on vibration. The vibration pick ups were located at the pilot's seat (sta 46.7) and in the cargo area at station 133. The vibration data are presented in figures 102 through 111, appendix I.

Qualitative pilot comments indicated the vibration levels in the UH-1N were less than those in the UH-1F. Vibration levels were highest at light gross weights at V_{max}. However, at airspeeds below 120 KIAS the vibration level was well within the comfort range. Qualitative analysis indicated that higher vibration levels were present with the aft cg configuration. For most conditions, the UH-1N met the specification of MIL-H-8501A (reference 3).

Lateral

In general, the lateral vibrations at station 133 were greater than at the pilot's seat for the first harmonic (2/Rev), but for the second (4/Rev) and third (6/Rev) harmonics, the vibrations at the pilot's seat were greater than at station 133. When the coefficient of thrust (C_T) was held constant, the higher rotor speed produced the lower vibration levels. In the aft cg loading, the vibrations at the pilot's seat were greater than with the forward cg loading; vibrations at station 133 remained approximately constant.

Vertical

In general, for vertical vibrations the first harmonic had the higher vibration levels at the higher airspeeds and were greater at the pilot's seat than at station 133. For the second and third harmonics, the higher vibration levels were at the lower airspeeds and were greater at station 133 than at the pilot's seat. For the constant C_T condition, the higher rotor speeds had the lower vibration levels. With the aft cg loading, the vibrations at the pilot's seat were greater than with the forward cg loading, while the station 133 vibrations remained approximately constant.

Pitch Link Loads

Pitch link load data were collected throughout the Category II performance program. One pitch link on the main rotor was instrumented by BHC to determine actual pitch link loads during flight. Pitch link load data were collected during all flight modes (level flight, partial power descent, autorotation, climb, landing, and takeoff), and throughout the flight envelope. Only level flight data were reduced for analysis; however, time histories of other flight modes illustrating typical wave forms are shown in figure 22.

Pitch link load data are presented in terms of the range load curve. The range load is defined as follows:

$$\text{Range Load} = \frac{\text{Maximum Load} - \text{Minimum Load}}{2}$$

where the maximum load and the minimum load are determined during one revolution of the main rotor.

Figure 112, appendix I, illustrates the variation of range load with μ, V_T, and C_T. The three flights presented were at a constant referred rotor speed of 320 with C_T varying from 32.0×10^{-4} to 50.0×10^{-4}. The general shape of the curves resembles that of a speed-power plot in that the variation of range loads with airspeed exhibits a "bucket." There was no apparent "knee" (drag divergence) in the range load curves as was experienced with the CH-47 (FTC-TR-66-46, reference 4). It should be noted that the CH-47 has a fully articulated rotor system, whereas the UH-1N has a semi-rigid teetering main rotor system. The minimum level of range loads increased by approximately 100 pounds as C_T increased from 32.0×10^{-4} to 50.0×10^{-4}. At airspeeds above 85 knots the range loads at high values of C_T increased more rapidly than did the range loads at lower values of C_T.

Figure 113, appendix I, presents a family of curves at a constant C_T of 43 x 10^{-4} for values of referred rotor speeds varying from 300 to 340. As in figure 112, appendix I, the general shape of the curves resembles a speed-power plot. At the lower referred rotor speeds (300, 310, 320) there was an indication of a "step" input in the "bucket" of the curve. The presence of this anomaly was noted in the speed-power data during this program as well as other helicopter performance programs conducted at the AFFTC. Indications of this "step" were not as pronounced for the higher referred rotor speeds (330 and 340). As referred rotor speed decreased from 340 to 310, the range loads also decreased. However, as the referred rotor speed was reduced from 310 to 300, the range load increased instead of decreasing as might have been expected.

Actual values of pitch link loads recorded during level flight varied from approximately 1,375 pounds as a maximum tensile load to approximately 425 pounds as a maximum compression load. These maximum values occurred on separate flights. The largest pitch link load variation observed during any one rotor revolution was approximately 1,200 pounds.

At airspeeds between approximately 117 KCAS and 45 KCAS the maximum tensile loads always occurred during the retreating portion of the blade cycle, and the lowest tensile loads (maximum compression load) occurred on the advancing segment of the revolution. However, as airspeed decreased below 45 KCAS, the point at which the maximum tensile load occurred shifted from the retreating phase to the advancing phase of the revolution. The point of occurrence of the minimum tensile load (maximum compression load) did not shift. Similarly, at airspeeds above 117 KCAS the point at which the minimum tensile load (maximum compression load) occurred shifted from the advancing segment of the revolution to the retreating portion of the blade cycle. The point of occurrence of the maximum tensile load did not shift.

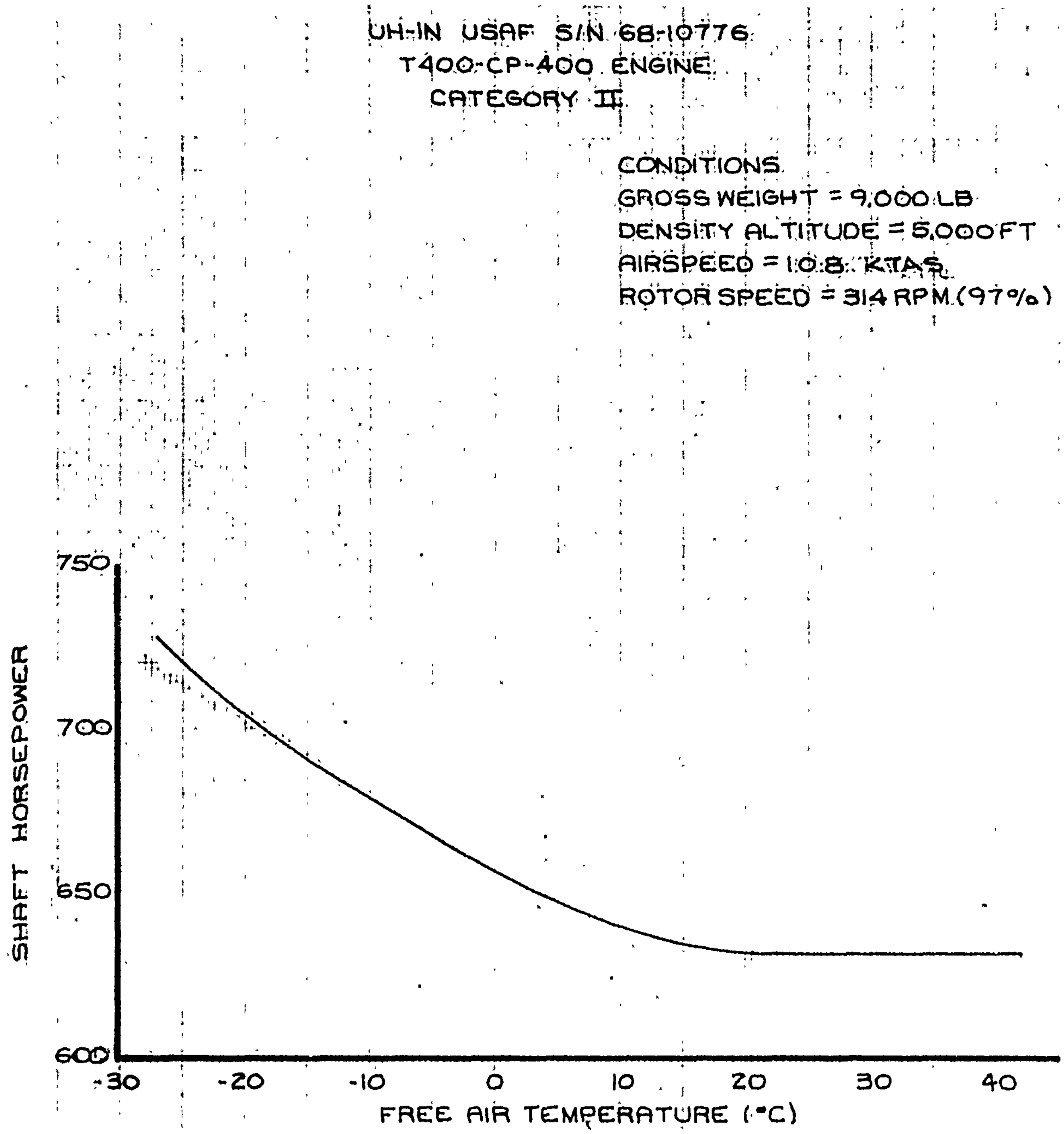


FIGURE 8. COMPRESSIBILITY EFFECTS ON POWER REQUIRED FOR LEVEL FLIGHT

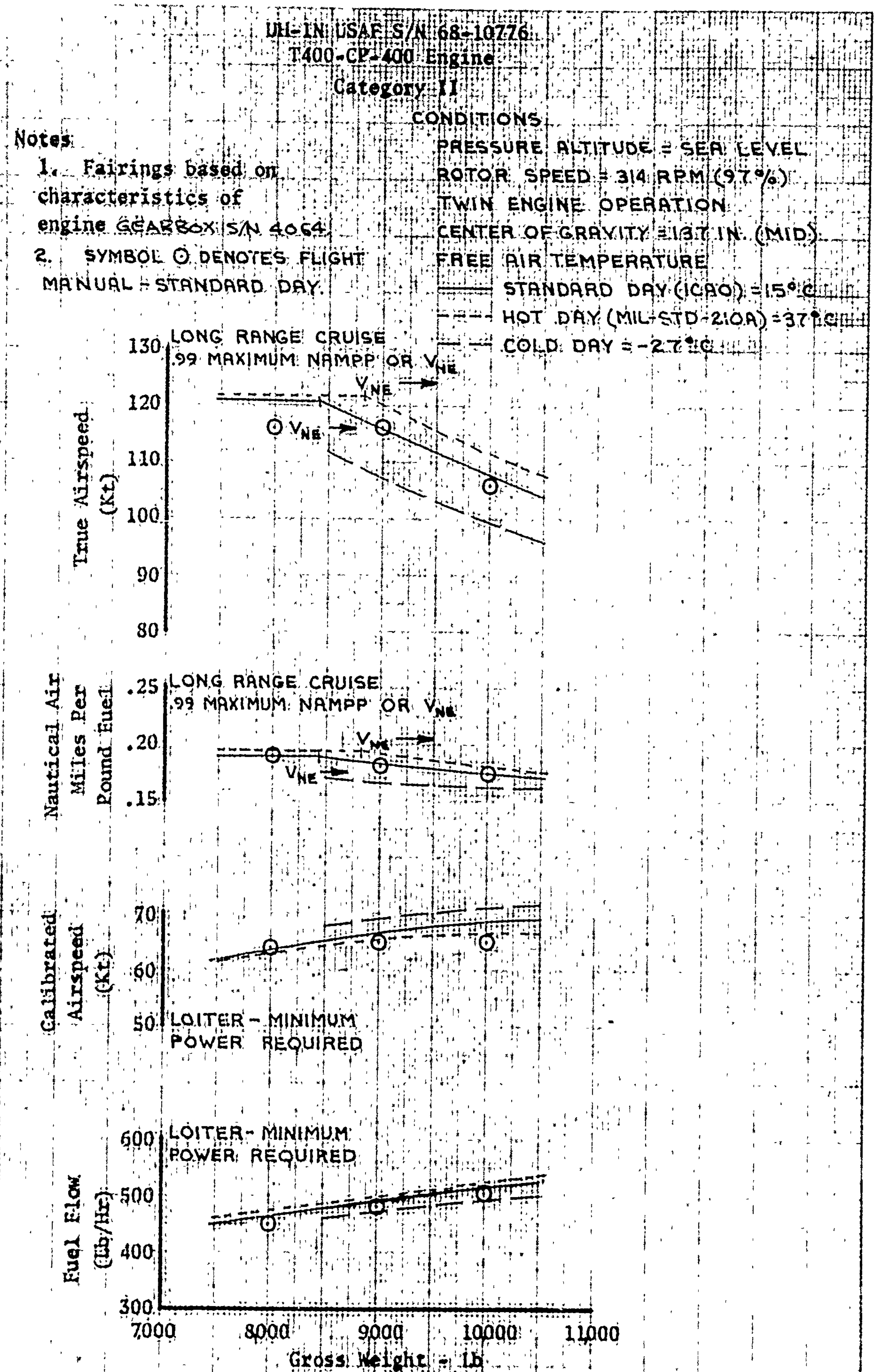


Figure 9. Level Flight Performance Summary - Twin Engine

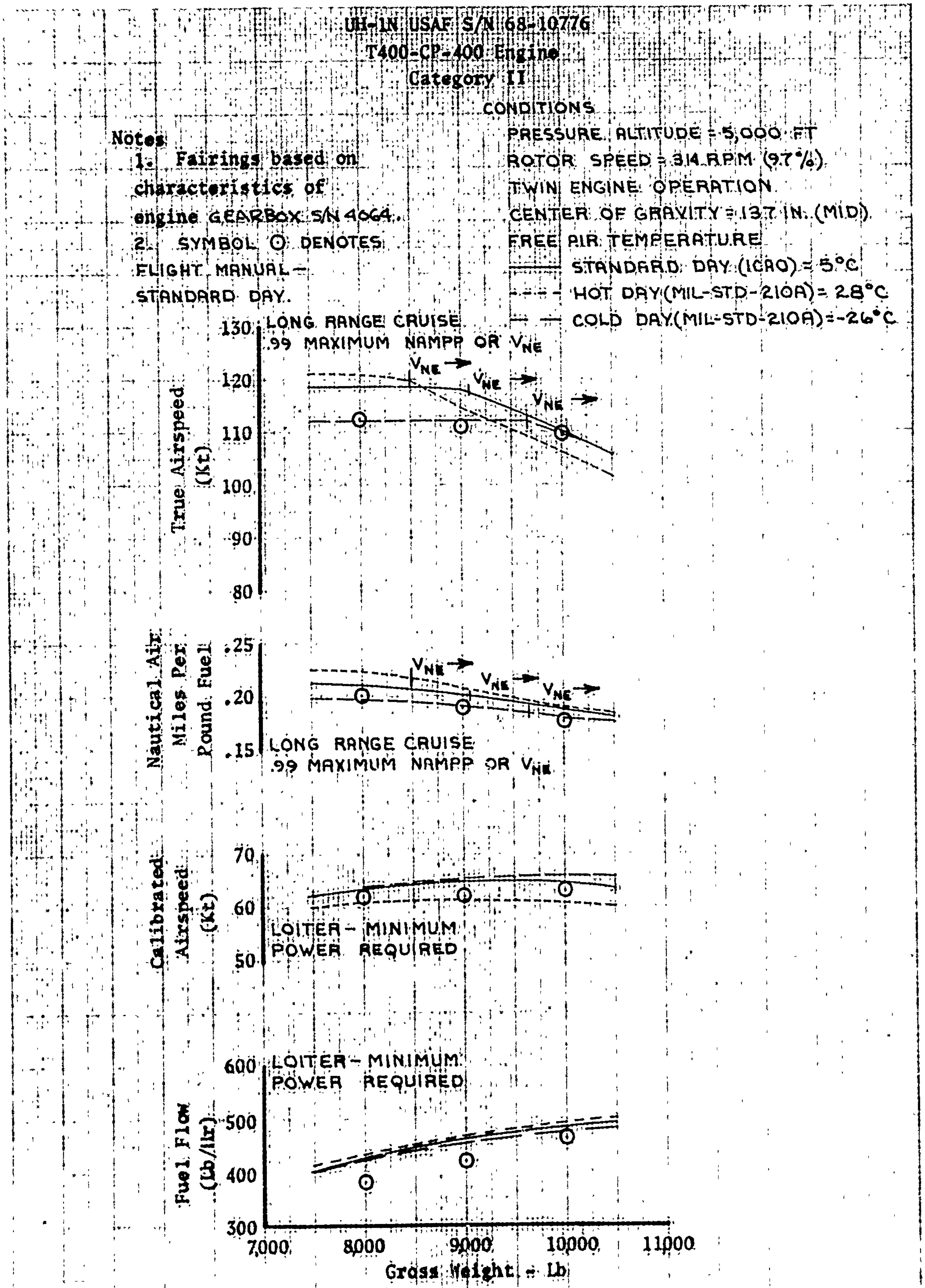


Figure 10. Level Flight Performance Summary - Twin Engine

UH-1N USAF S/N 68-10776
T400-CP-400 Engine
Category II

Notes
1. Fairings based on characteristics of engine GEARBOX S/N 4064.
2. SYMBOL ⊙ DENOTES FLIGHT MANUAL - STANDARD DAY

CONDITIONS
PRESSURE ALTITUDE = 10,000 FT
ROTOR SPEED = 314 RPM (97%)
TWIN ENGINE OPERATION
CENTER OF GRAVITY = 137 IN. (MID)
FREE AIR TEMPERATURE
——— STANDARD DAY (ICAO) = -5°C
- - - - HOT DAY (MIL-STD-210A) = 18°C
— — COLD DAY (MIL-STD-210A) = -26°C

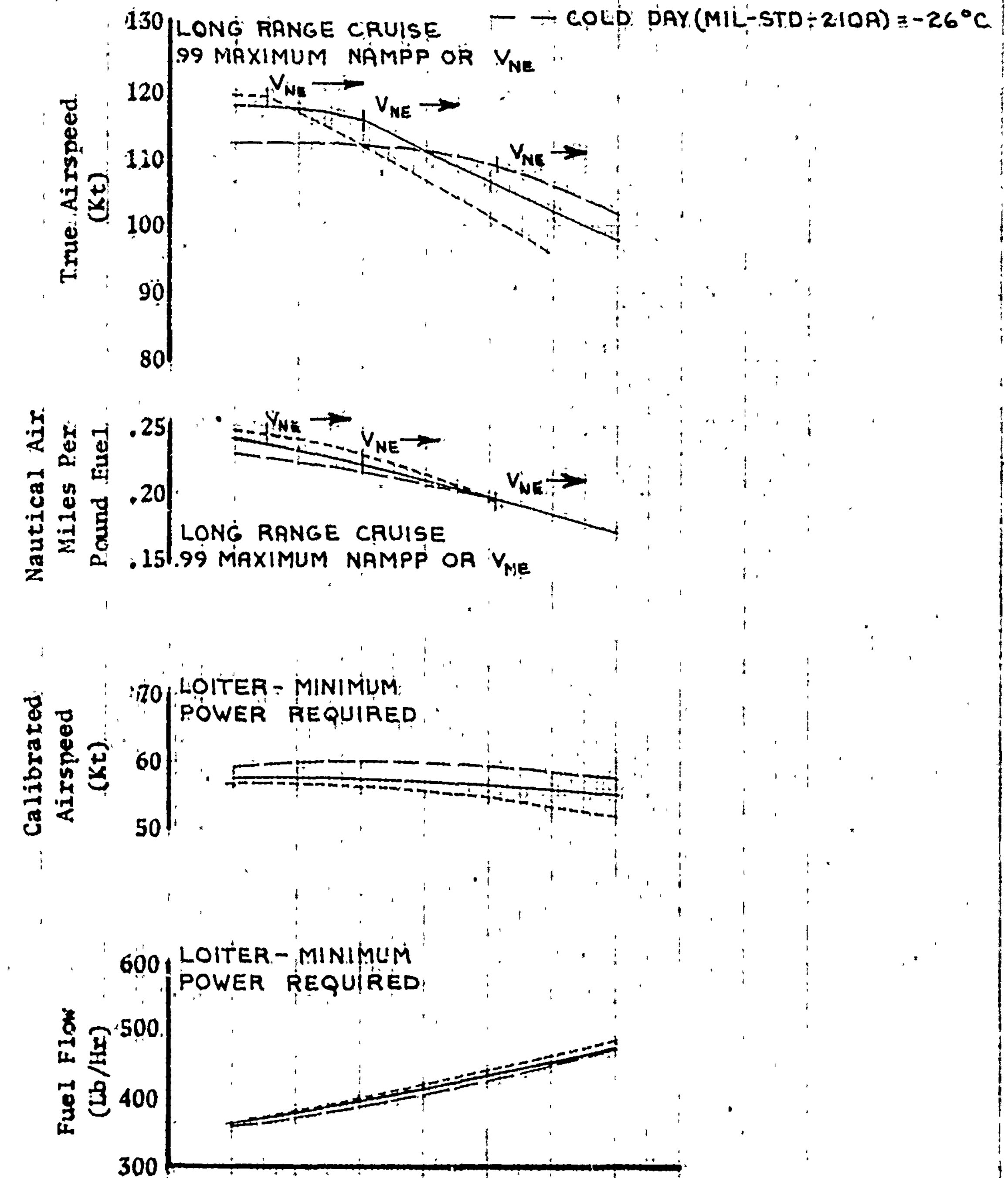


Figure 11. Level Flight Performance Summary - Twin Engine

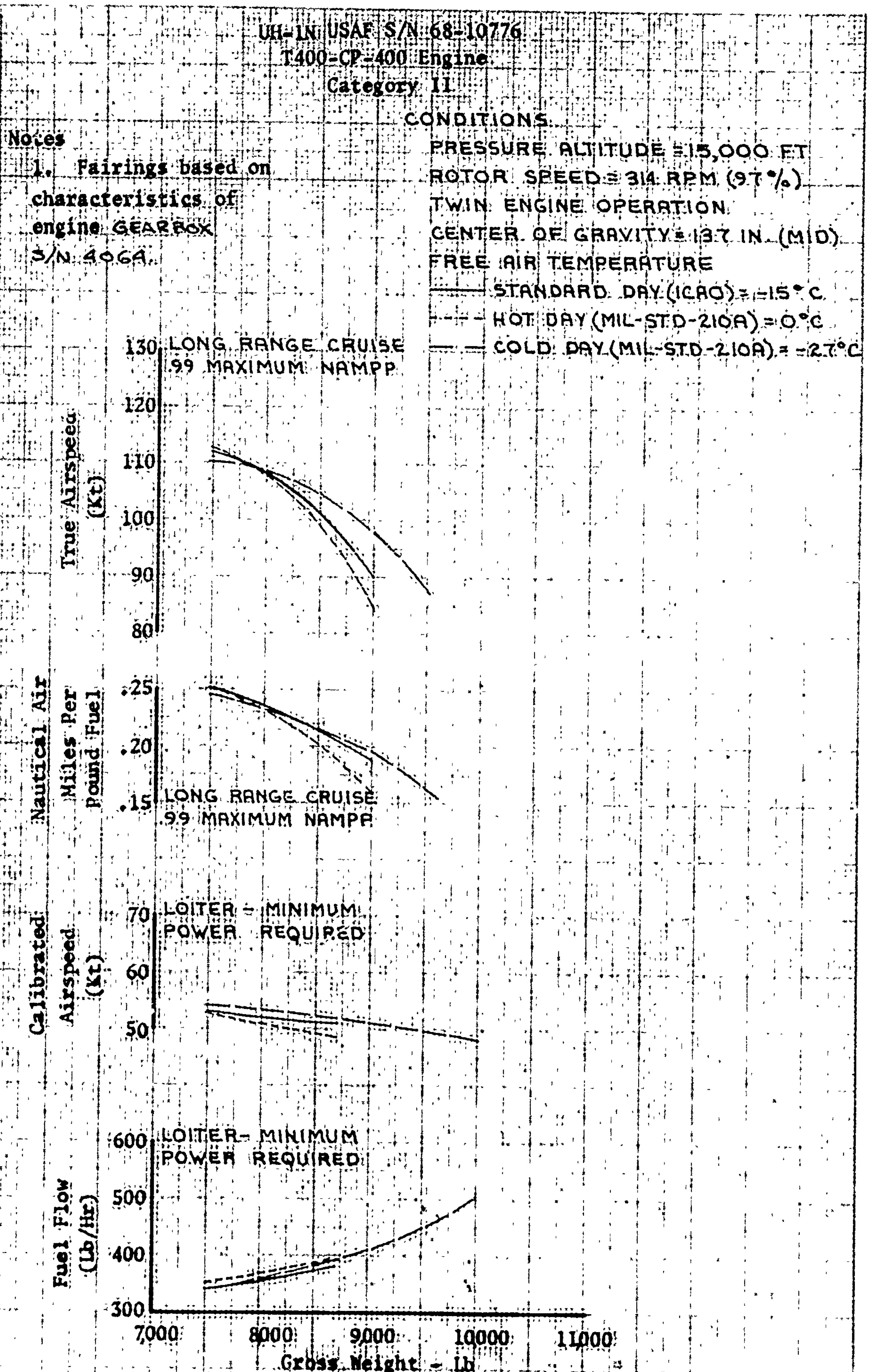


Figure 12. Level Flight Performance Summary - Twin Engine

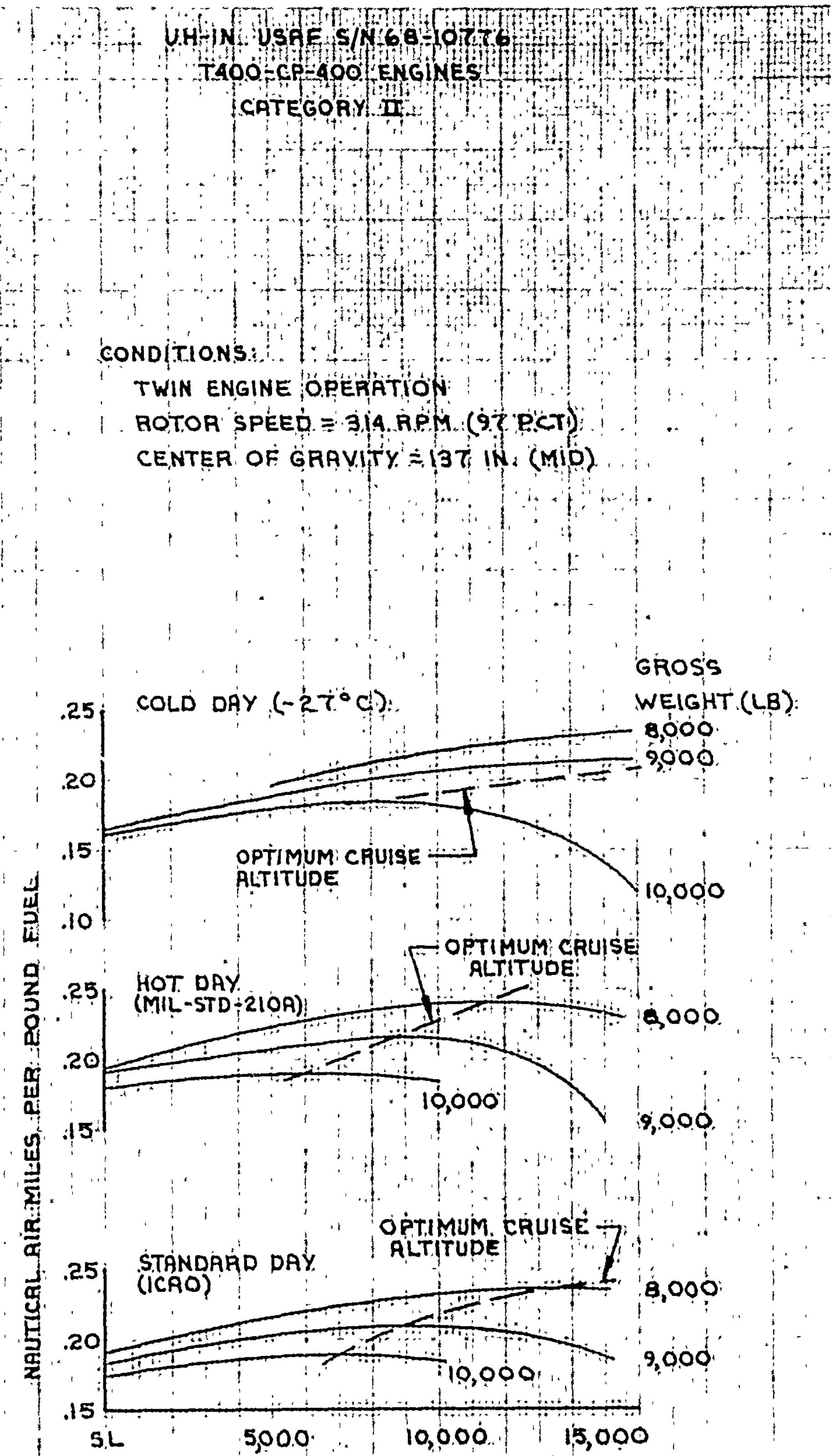


FIGURE 13. OPTIMUM LEVEL FLIGHT MAXIMUM RANGE CRUISE ALTITUDE SUMMARY

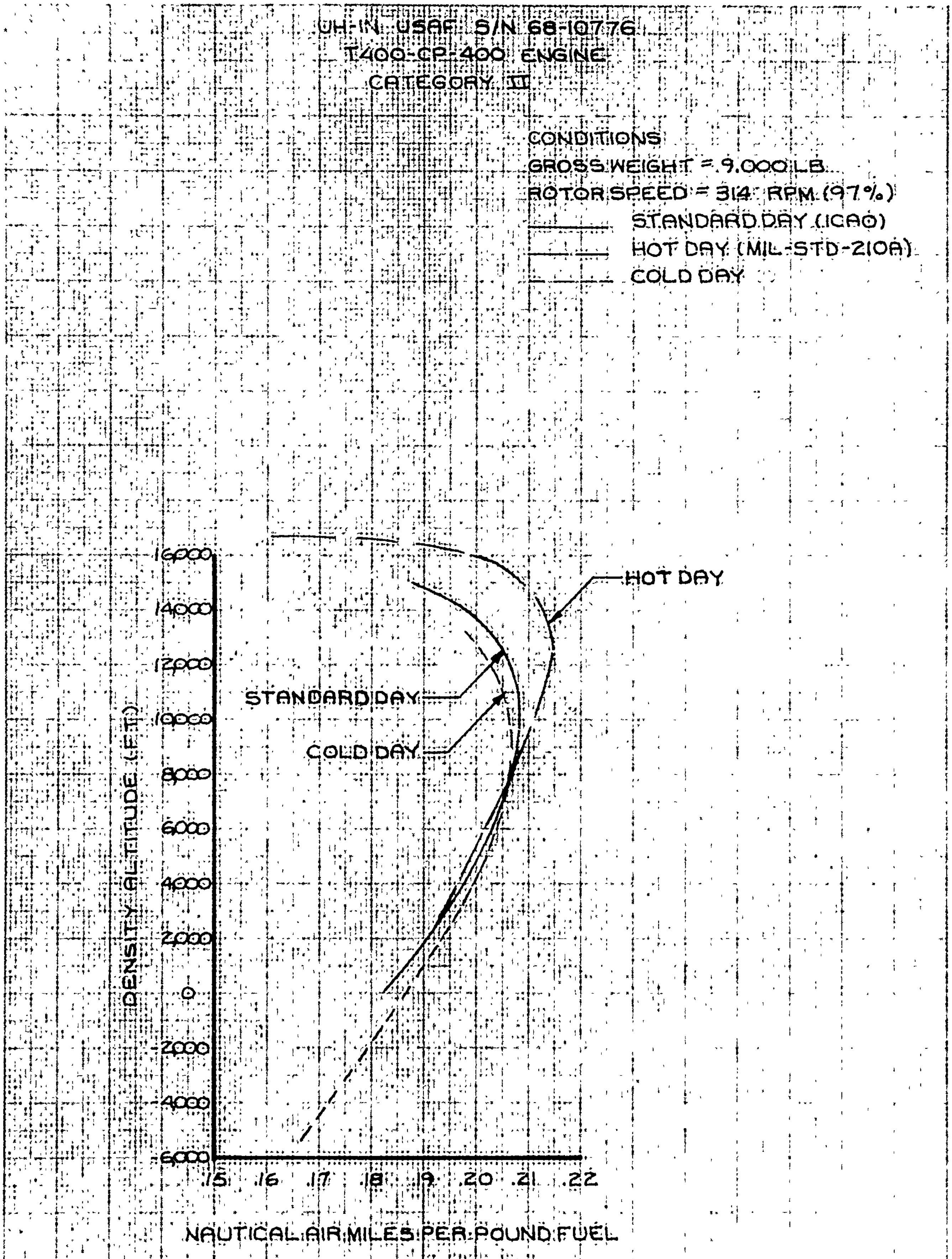


FIGURE 1A. MAXIMUM RANGE CRUISE PERFORMANCE ALTITUDE EFFECTS SUMMARY

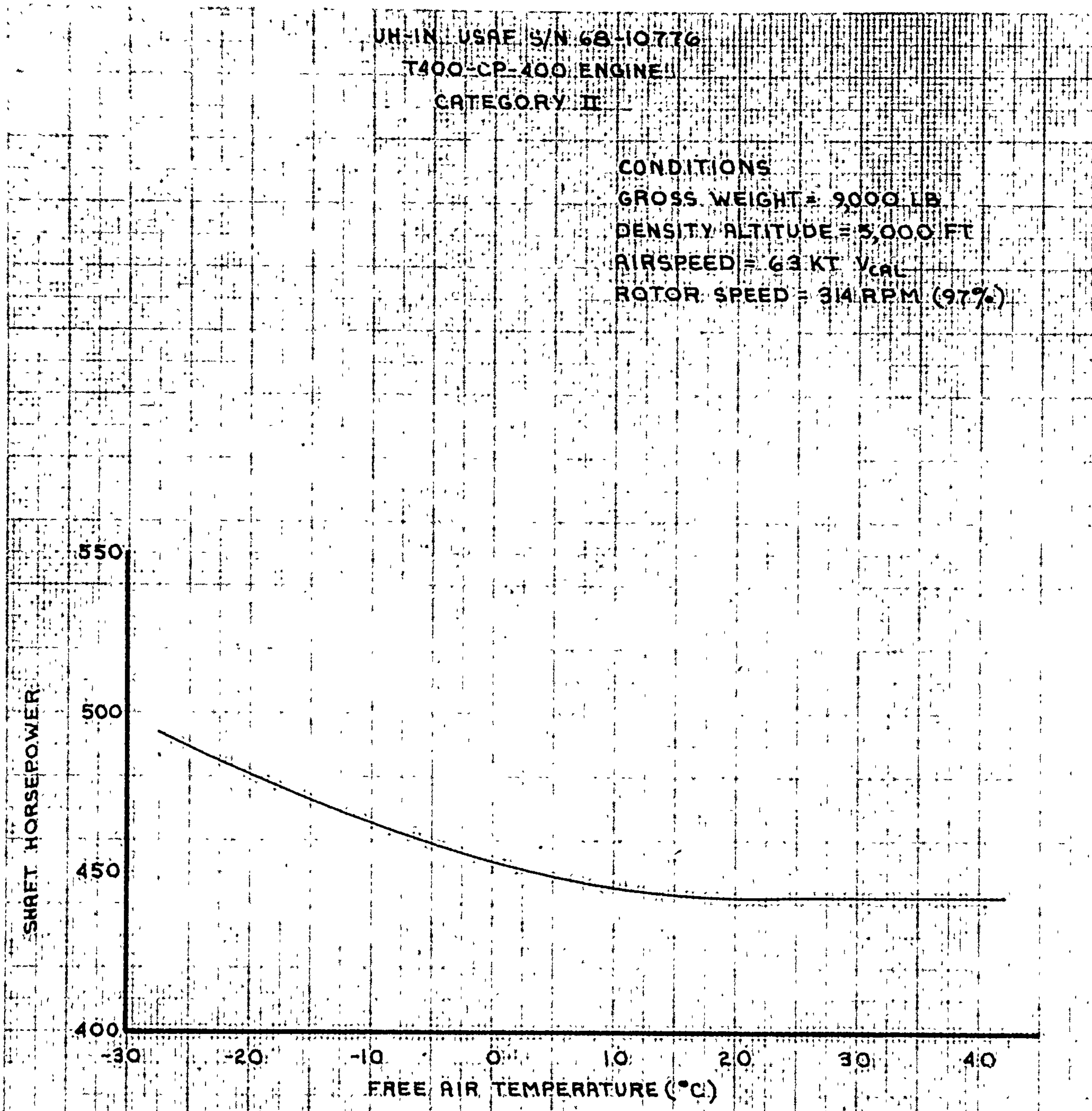


FIGURE 15. COMPRESSIBILITY EFFECTS ON MAXIMUM ENDURANCE POWER REQUIRED

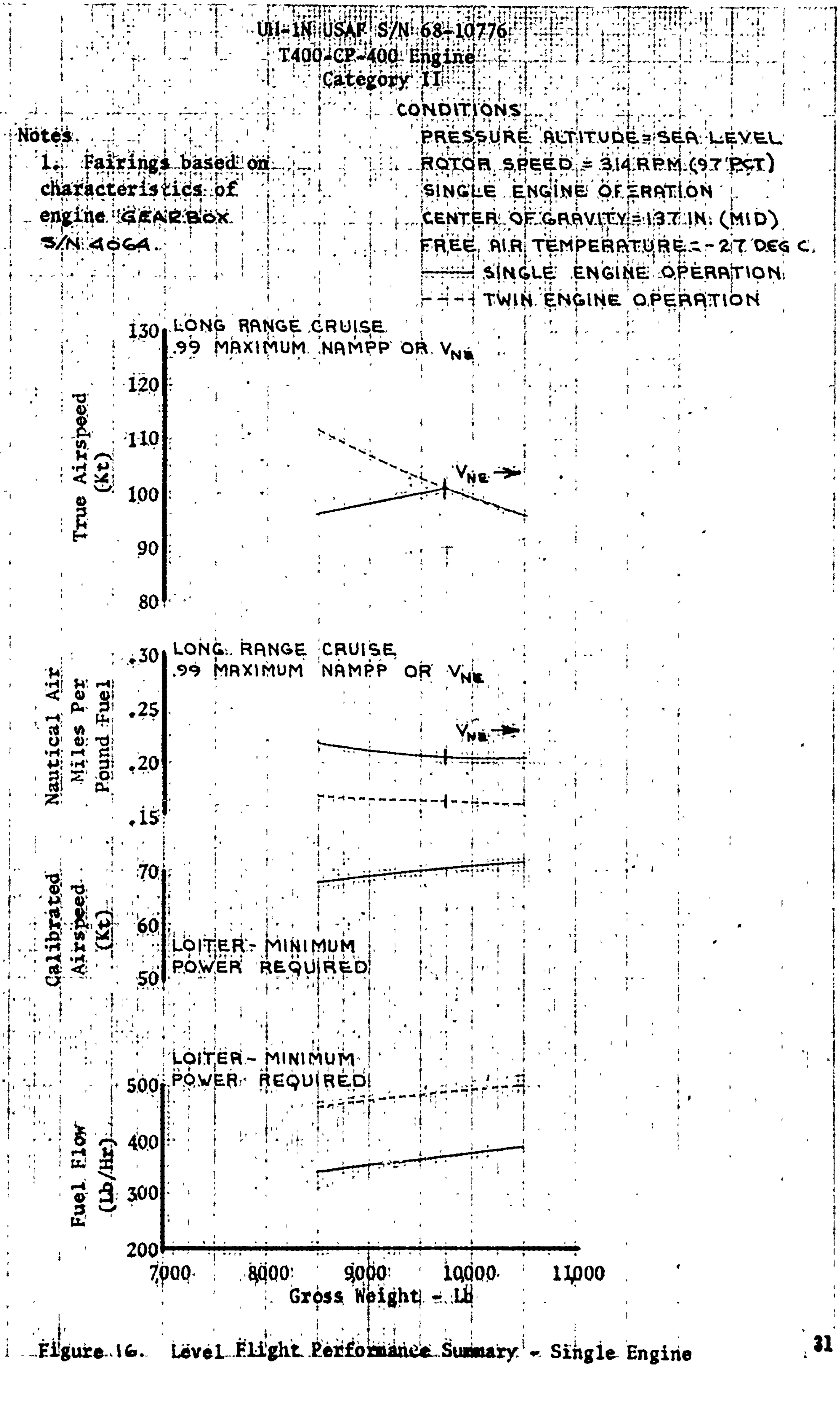


Figure 16. Level Flight Performance Summary - Single Engine

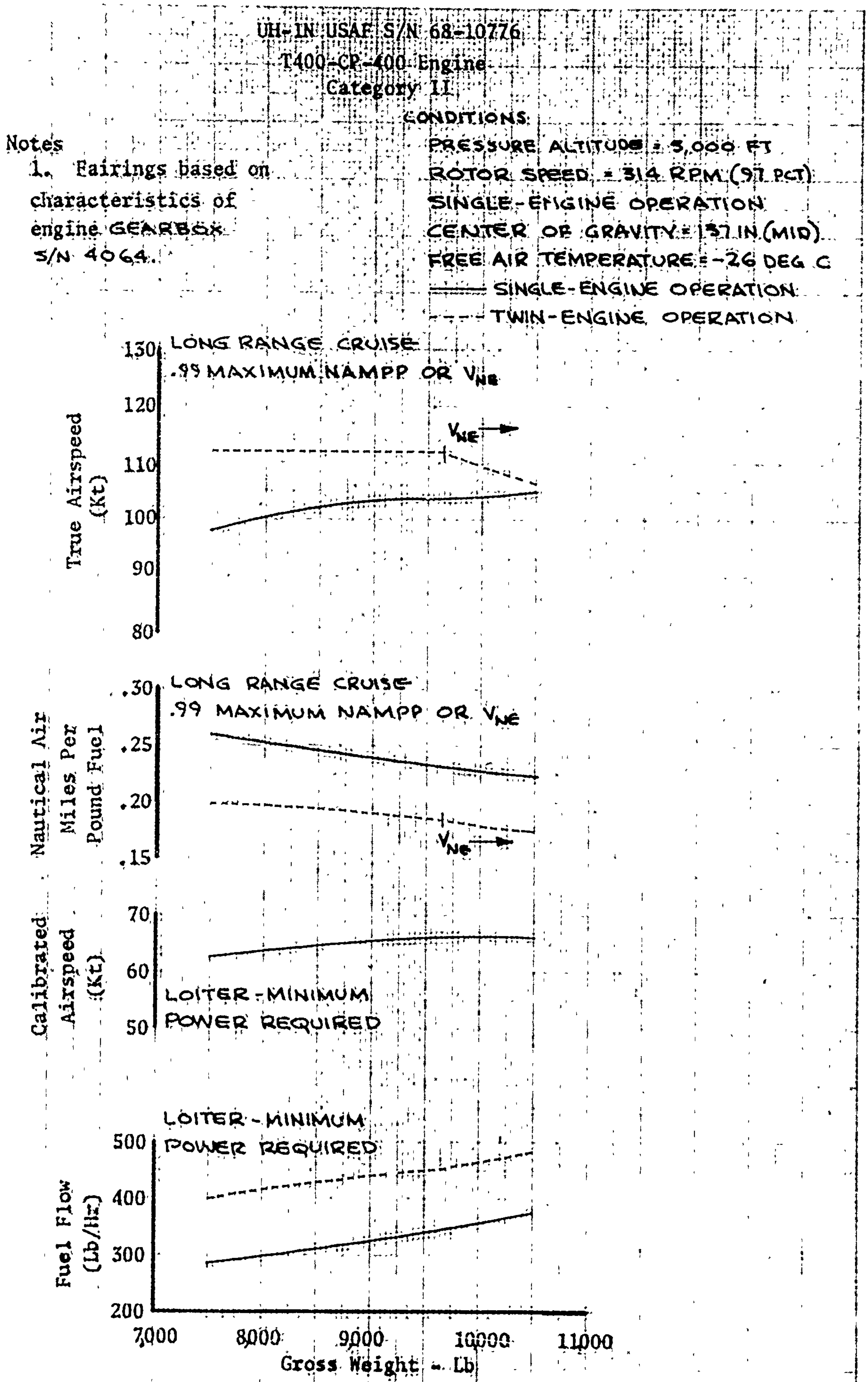


Figure 17 Level Flight Performance Summary - Single Engine

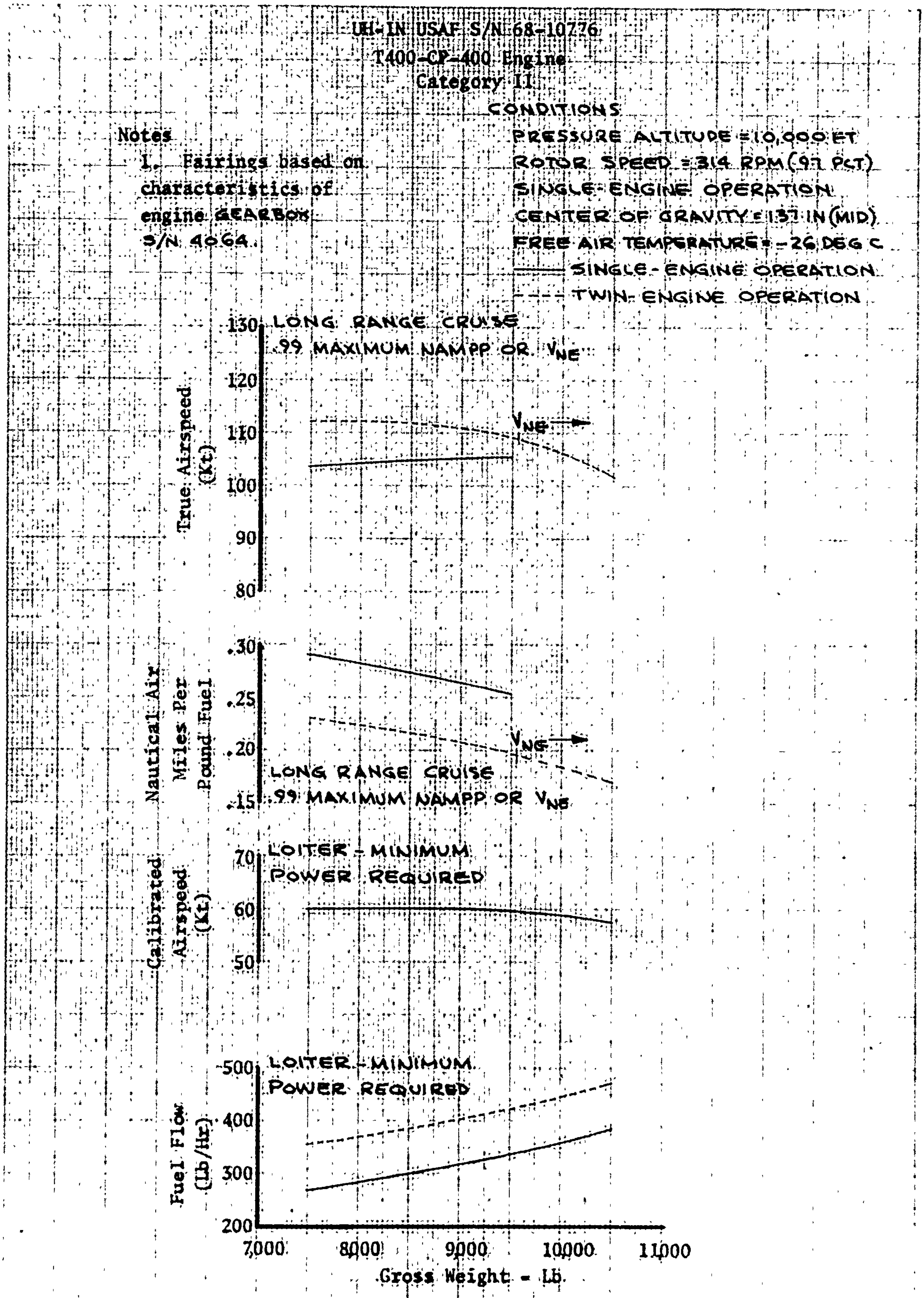


Figure 18 Level Flight Performance Summary - Single Engine

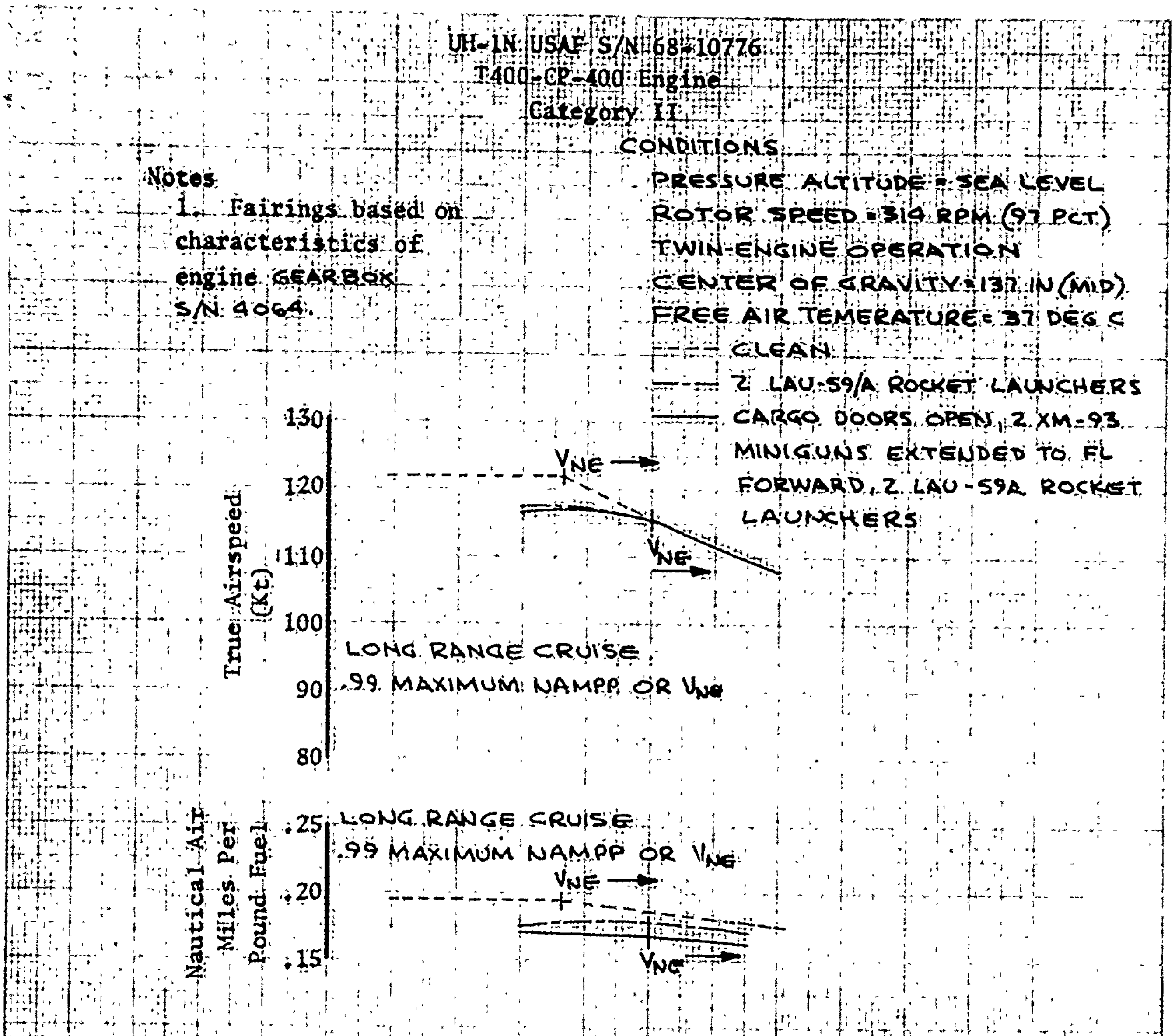


Figure 19 Level Flight Performance Summary - With External Armament

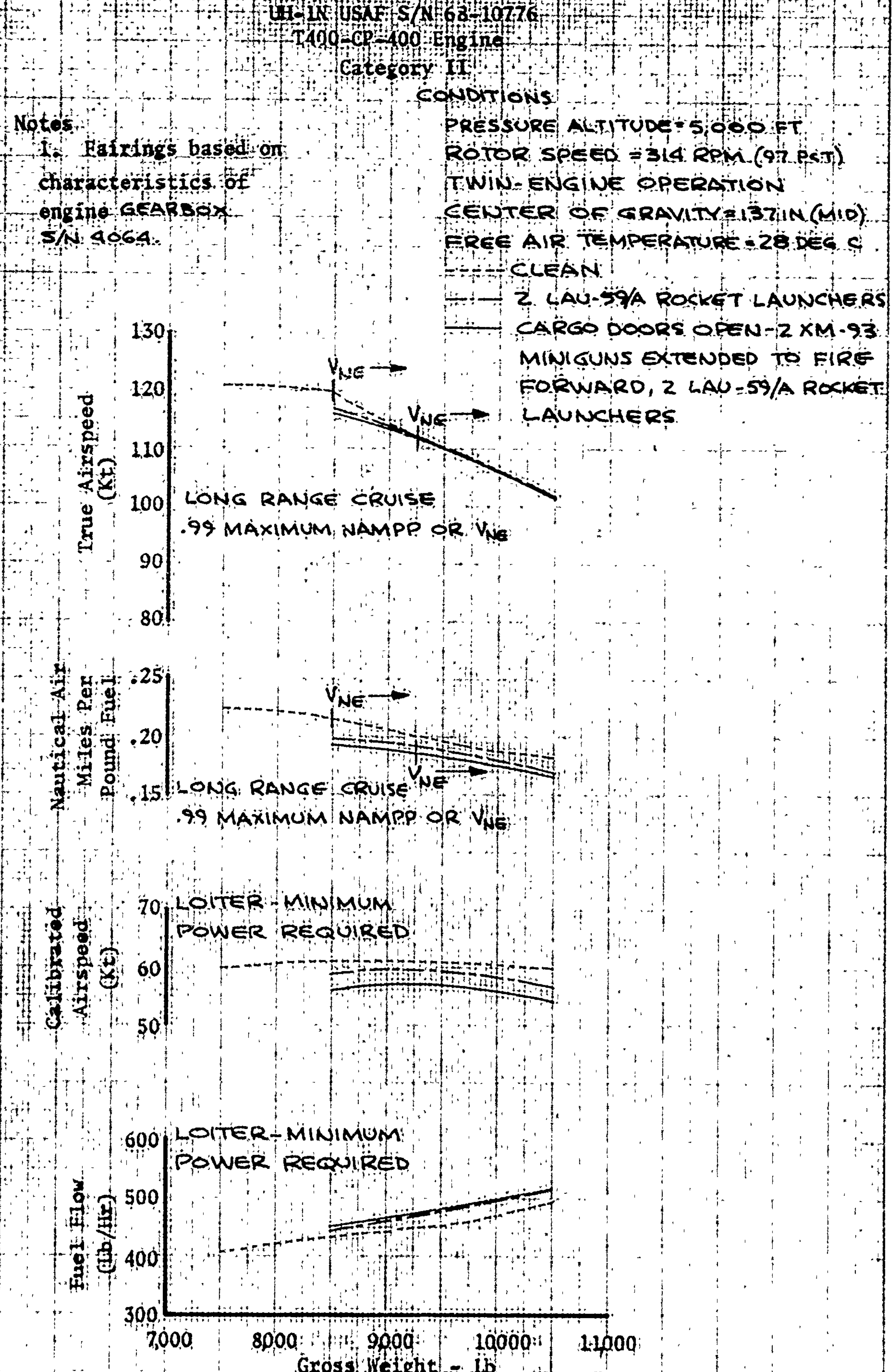


Figure 20 Level Flight Performance Summary - With External Armament

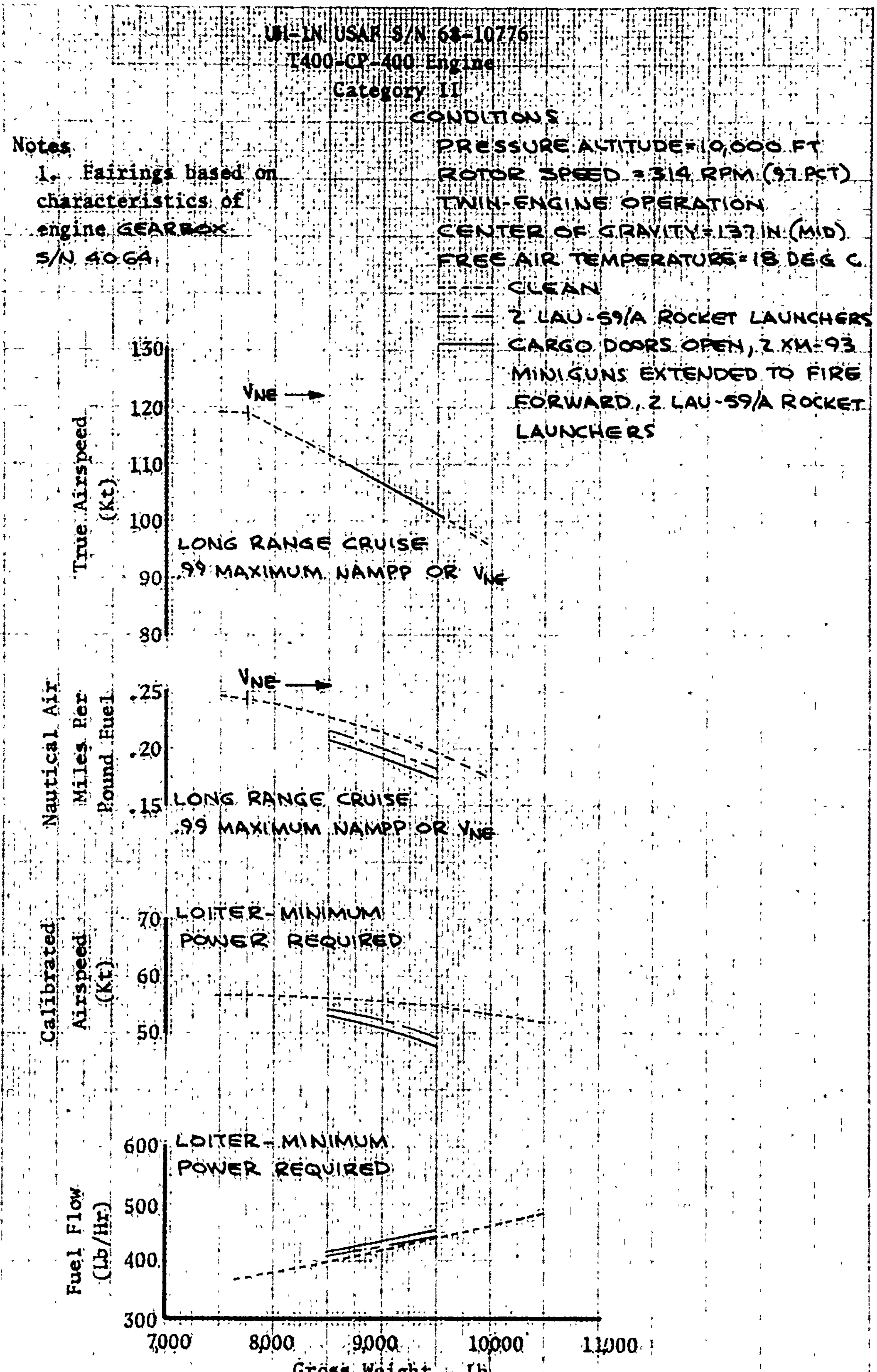
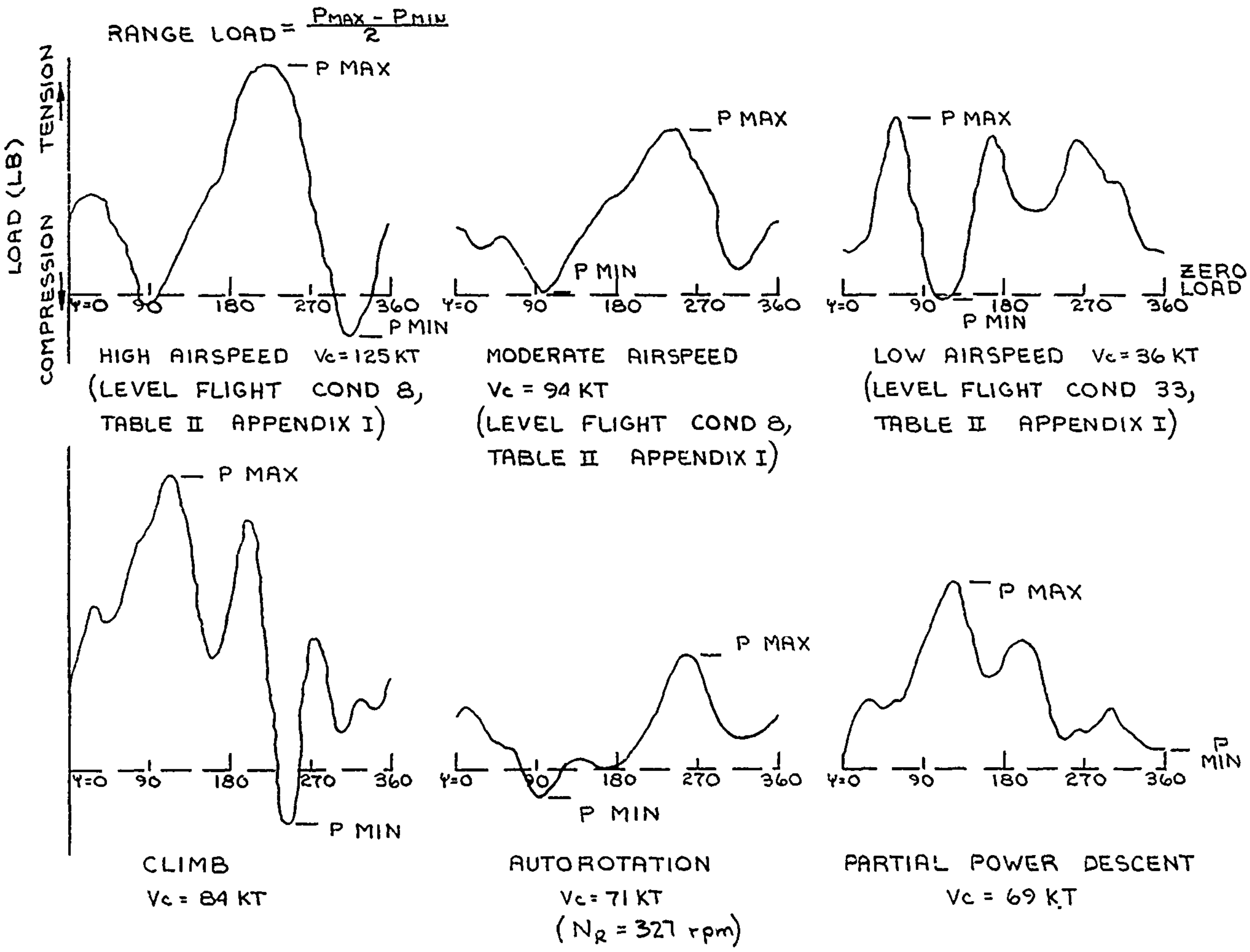


Figure 21 Level Flight Performance Summary - With External Armament



NOTES:

1. THE CLIMB, AUTOROTATION, AND PARTIAL POWER DESCENT DATA ARE FOR AN AVERAGE 9,900 LB GROSS WEIGHT, 9,200 FT PRESSURE ALTITUDE, 6 DEG C AMBIENT TEMPERATURE AND MID-CG LOCATION.
2. THE DIRECTION OF ROTATION IS COUNTERCLOCKWISE WITH Ψ = 0 BEING THE AZIMUTH POSITION OVER THE TAIL OF THE HELICOPTER.

FIGURE 22 PITCH LINK LOADS SURVEY

Autorotational Descent Performance

Autorotational descent performance tests were conducted at gross weights of 8,500 and 10,000 pounds and at pressure altitudes of 5,000 and 10,000 feet. Rotor speeds from 294 rpm (91 percent) to 339 rpm (104.5 percent) were tested at various airspeeds. The results of the autorotational descent tests are presented in figures 114 through 117, appendix I.

The tests showed that relatively little change in rate of descent occurred with changes in weight or altitude. Increases up to 30 feet per minute rate of descent occurred as gross weight increased from 8,500 to 10,000 pounds. For a given gross weight, the rate of descent increased approximately 100 feet per minute when the altitude was increased from 5,000 to 10,000 feet.

Rotor speed had the greatest effect on the rate of descent, with the minimum allowable rotor speed of 294 rpm (91 percent) resulting in the lowest rate of descent. At a rotor speed of 294 rpm the minimum rates of descent were approximately 1,620 and 1,720 feet per minute at pressure altitudes of 5,000 and 10,000 feet, respectively. For airspeeds near the minimum rate of descent, the rate of descent increased approximately 325 feet per minute as the rotor speed was increased from 294 to 339 rpm.

The calibrated airspeeds for minimum rate of descent and maximum glide distance decreased slightly with rotor speed (2 to 3 knots), but were essentially independent of gross weight and altitude. For a rotor speed of 294 rpm the calibrated airspeeds for minimum rate of descent and maximum glide distance were found to be 66 and 86 KCAS, respectively.

Slope Landing

Slope landings were made to determine the slope angles upon which the UH-1N could be safely landed with various gross weights and cg locations. Before starting the actual slope landings, the main-rotor-to-fuselage clearance was investigated with various fore and aft cyclic control and collective control inputs.

The fuselage-rotor blade clearance tests showed that the rotor blades at no time came closer to any portion of the fuselage than 10 to 12 inches. Simultaneous aft movement of the cyclic and lowering of the collective produced this minimum clearance from the tailboom. To produce this situation the cyclic was moved 6.3 inches aft to the stop and the collective was lowered 4.1 inches to full down, all in 0.3 second. Simultaneously lowering the collective 4.1 inches to full down and forward motion of the cyclic 6.4 inches to full forward, within 0.3 second, gave 15 to 20 inches clearance from the test noseboom. The nearest the rotor blade came to any standard installation on the forward section of the aircraft was approximately 15 inches from the UHF/VHF antenna located above the pilots' compartment. No overrunning of the stabilized position of the rotor blade was evident when the rotor was unloaded by rapidly dropping the collective.

The slope landings were accomplished on semi-prepared surfaces on a hill having a wide variety of slope angles up to approximately 17 degrees.

The surface was typical of those found in this desert region - decomposed granite and irregular quartz rock ranging in size from very fine gravel to rocks as large as 3 inches in diameter. The helicopter landing skids made slight (if any) imprint on the surface. This surface was relatively slippery at the higher slope angles and required care when landing the aircraft. The nature of the surface did not limit the slope angles attained, however. The maximum angles attained were dictated by cyclic control limits or structure-ground clearance. These tests were conducted using the techniques outlined in the Flight Manual, and these techniques were satisfactory under the conditions tested. The results of the slope landing tests are presented in figure 118, appendix I.

Cross-slope maximums were dictated by lateral cyclic control limits. The tests were conducted without limiting the control movement to a 10-percent-remaining range. Therefore, the slope angles allowing a 10-percent remaining-control margin were slightly less than the maximum slopes obtained during the tests.

Nose up-slope maximum points were affected by the test noseboom installation configuration more than forward cyclic control limits. A nose up-slope maximum of 17 degrees accommodated all permissible longitudinal cg locations which allowed at least a 10-percent-remaining control margin. At this slope angle, the mid cg condition control margin was approximately 10 percent, while with a forward cg condition, fuselage proximity to the ground was the limiting factor. With an aft cg condition 20-percent longitudinal control margin remained.

Nose down-slope angles were limited solely by the tail skid proximity to the ground. Mid and forward cg conditions resulted in essentially the same slope angles of 10 and 11 degrees respectively. An aft cg condition resulted in a greatly reduced maximum slope of 5 degrees. More than 10-percent control margin remained at the maximum slope angles for all the cg conditions.

Height-Velocity

Tests were conducted to determine the envelopes which, in case of a single-engine failure, defined the minimum height-velocity combinations from which flight could be maintained or a safe landing effected. The approach to the conduct of these tests was to maintain maximum available power on the remaining engine during all phases of the landing or airspeed recovery to effect a go-around.

The UH-1N exhibited excellent single-engine performance at relatively low height-velocity combinations. Engine response time was sufficiently rapid to prevent large rotor speed losses when one engine was retarded; consequently, large reductions in collective control to regain rotor speed were not necessary. Maximum available power on the remaining engine was easily attained and maintained. The single-engine height-velocity test results are presented in figures 119 through 126, appendix I.

The power ratio (single-engine power available divided by the power required to hover OGE) was a critical factor in the height-velocity performance; that is, when the density altitude was reduced, more power was available and less power was required to hover OGE; consequently, a higher

power ratio resulted. For example, at a gross weight of 9,990 pounds and density altitude 4,730 feet, a power ratio of 0.6100 resulted; however, at 10,500 pounds and density altitude of 3,160 feet, a power ratio of 0.6435 developed which resulted in slightly reduced AVOID and CAUTION areas (reference figures 121 and 122, appendix I).

The following conditions tested did not result in height-velocity curves.

Gross Weight (lb)	Pressure Altitude (ft)	FAT (deg C)	Density Altitude (ft)	Power Ratio
7,500	3,890	11	4,310	725/690 = 1.051
8,410	2,040	4.7	1,380	889/876 = 1.014

Single-engine chops were made from a hover at all skid heights from 300 feet down to as low as 3 to 4 feet without striking the ground or having to land. In both cases, the power ratio was greater than one.

The single-engine GO-AROUND area encompasses those test points in which the go-around minimum height above the ground was in excess of 5 feet and no difficulties were encountered. The CAUTION areas contain the marginal go-arounds in which the height above the ground was less than 5 feet, or those wherein very poor acceleration to climb speed was encountered, and those landings which could have been a go-around instead of a landing. The boundary that delineated the AVOID area was determined by mandatory landings. In general, the UH-1N could be consistently landed at true airspeeds near or less than the 15 knots specified in MIL-H-8501A (reference 3). At 9,500 pounds gross weight and a density altitude of approximately 10,700 feet, the minimum landing speeds were not more than 19 knots true airspeed (KTAS).

The following discussion should be included in the Flight Manual after the first paragraph under the heading FLIGHT CHARACTERISTICS UNDER SINGLE ENGINE CONDITIONS (in Section II under ENGINE FAILURE): **(R 4)**

FLIGHT OPERATION NEAR THE AVOID AREA

The failure of an engine near the AVOID area of the height-velocity diagram requires prompt action by the pilot if a safe landing or a go-around is to be made. When operating near the AVOID area, the pilot should be aware of his minimum single engine level flight speed and climb speed. These speeds will vary with helicopter gross weight and density altitude. If altitude permits, at least a 20° nose-down attitude should be established to accelerate to the level flight or go-around airspeed. At the same time, the power on the operating engine should be increased to maximum and collective pitch should be used to establish 97% N_R minimum. As the height at which an engine failure occurs decreases, a progressively shallower nosedown attitude should be used. Below 30 feet the collective pitch should be

lowered only slightly to regain rotor rpm to avoid building up a high sink rate. As the level flight airspeed is reached, the helicopter should be returned to a level attitude and a climb established after climb speed is attained. When clear of all obstacles, the aircraft should be accelerated to above 55 KIAS.

If a landing is required following the loss of an engine, two techniques may be used, depending on the landing speed required. If a prepared surface is available, a single engine slide landing can be made using a skids-level attitude and collective pitch to cushion the landing. If a slow touchdown speed is required, a moderate flare can be used at 25 feet to slow the helicopter. Maintain maximum power on the operating engine in the flare by maintaining rotor rpm at 97% N_R. An increase in rotor speed in the flare may cause the operating engine N_f governor to sense an overspeed and cause a reduction in power. Establish a skids level attitude prior to touchdown and cushion the landing with collective pitch.

The following discussion should be included in Appendix I, Flight Manual Performance Data, under the paragraph titled: Height Velocity Chart. **(R4)**

The height velocity diagrams are plots of minimum heights versus airspeed for a safe single engine landing and/or go-around. The curves obtained are based on level, unaccelerated flight, in very light wind conditions (less than 3 knots). The information is based on a rotor rpm of 100% prior to engine failure and maintaining 97% N_R during the descent, go-around or landing. No consideration was given to altitude loss due to nonstandard pilot technique and turns. The green area of each curve represents the altitude/airspeed combinations in which a single engine go-around can be made. The yellow area represents the altitude/speed combinations in which a safe landing can be made; and the red area indicates those combinations in which the loss of an engine will most probably result in damage to the helicopter. Continuous flight at altitude/airspeed combinations within the red or yellow areas of the curves should be avoided.

Engine Performance

General

Steady state engine, engine inlet, and engine power available (topping power) data were obtained over a wide range of ambient temperatures and pressure altitudes (-38 degrees C to +42 degrees C and sea level to 15,000 feet H_p).

These data (figures 127 through 166, appendix I) were primarily collected using UH-1N S/N 68-10776 (referred to as 776); however, topping power data collected on UH-1N S/N 68-10774 during testing in the climatic laboratory at Eglin AFB, Florida, are included.

Two calibrated T400 engine packages were required for use on 776 during the test program; they were gearbox S/N 4064 with power sections S/N 66127 and S/N 66128, and gearbox S/N 4061 with power sections S/N 66126 and S/N 66122. Power section S/N 66121 was initially part of the S/N 4061 power package, but was replaced after a short period of time due to a damaged ITT thermocouple lead. Engine package gearbox S/N 4064 was replaced with package S/N 4061 after deterioration of power available became pronounced and reliability of the engine package decreased.

Engine response to transient power inputs was tested as part of the Category II Systems Evaluation Test Program, and the results are presented in reference 5.

Engine Inlet Performance

Engine inlet conditions (pressure and temperature) were determined at the inlet screen of each engine. Nine total temperature and three total pressure probes were stationed in series around each inlet screen to record average inlet values. The inlet data are shown in figures 139 through 141 and 154 through 159, appendix I.

Observation of the inlet characteristics indicated the following:

1. At power settings below 91-percent N_g, compressor inlet temperatures can rise 15 to 20 degrees C due to reingestion of compressor bleed air. This problem was previously identified in reference 2.

2. Rises in compressor inlet temperature at high power settings above 92-percent N_g typically averaged 3.5 degrees C in a hover, 5.0 degrees C in level flight and 6.0 degrees C in a climb.

3. The ratio of compressor inlet total pressure (P_{t_2}) to ambient pressure (P_a) typically averaged 1.002 in a hover, 0.996 in level flight, and 1.018 in a climb.

During the engine topping tests, it was noted that flight mode (level flight, hover, climb) did not influence the topping torque attained, that is, power available. It was therefore deduced that the inlet pressures and temperatures were independent of flight mode. The data presented in paragraphs 2 and 3 above showed there were inlet changes with respect to flight mode; however, calculation of power available using the inlet conditions for the various flight modes showed power available to be virtually the same for hover and climb, but for level flight it was approximately 11 shaft horsepower (shp) (1.0 percent) less than hover and climb for standard day sea level ambient conditions. Since the scatter bands of each set of data (temperature or pressure) for the various flight modes overlap each other, power available can be considered to be independent of flight mode. Inlet values recommended for calculating power available are a CIT rise of 5.0 degrees C and P_{t_2}/P_a of 1.005.

Power Available

Power available (topping power) was obtained using UH-1N's S/N 68-10774 and S/N 68-10776. The topping power presented for 774 was obtained during tethered ground run tests at Eglin AFB, Florida. Ambient temperatures for these tests were varied from -38 degrees to +42 degrees C. Almost all of the topping power presented for 776 was obtained during hover and level flight. Ambient conditions encountered during these tests were temperatures from -35 degrees to +32 degrees C, and pressure altitudes from 2,000 to 9,500 feet. The relationships of N_g, W_f, and ITT with OAT for the topping power checks are shown in figures 160 and 161, appendix I. The following observations can be made concerning these data:

1. The No. 1 (left) power sections topped only on N_g, except at cold temperatures when they topped on W_f. The No. 1 power sections were never observed to top on ITT.

2. The No. 2 (right) power sections generally topped on ITT, except at colder temperatures when they topped on N_g and W_f.

3. An average W_f limit of 575 pounds per hour was observed for both the No. 1 and No. 2 engines.

From the topping relationships, and using the single-engine relationships of figures 131 through 138 and 146 through 153, appendix I, the power available charts of figures 162 and 163, appendix I, were constructed.

ASD letter 12-72 (T400 Temperature Lapse Rate Data, Category II UH-1N Helicopter), presented expected (both U.S. Navy test and contractor estimated) power-available lapse rate data with respect to ambient temperature. Since power-available determinations have been a UH-1N problem, causing unwarranted engine changes in some cases, the above-mentioned letter requested AFFTC to provide as much information in this area as possible. Table VII shows the values determined by AFFTC flight test as compared to the other agencies. The differences between the No. 1 and No. 2 engines are apparent, and evidently a result of the parameter on which the engine topped, that is, either N_g or ITT. If the engine topped on ITT, the power-available difference from sea level standard day to +35 degrees C was a decrease of approximately 17 percent. If it topped on N_g, a power decrease of approximately 13 percent was noted. Power-available percentage increases due to a temperature change from standard day at sea level to -20 degrees C varied with each engine, because standard day power available changed and power available at -20 degrees C remained essentially constant at 900 shp. The decrease of power available per engine with pressure altitude was determined to be approximately 25 to 35 shp per 1,000 feet at pressure altitudes from sea level to 4,000 feet, and 20 to 25 shp per 1,000 feet for altitudes above 4,000 feet.

Calculated power available at sea level standard day for the four engines used in this test was 840, 825, 810 and 775 shp. Averaging these values and subtracting the average from 900 shp, yields 88 shp which represents an average installation loss for the UH-1N. This number compares favorably with the computed installation loss of 85 shp presented at the 1 December 1971 T400 engine meeting at NASC. The W_f limit of 575

pounds per hour was sufficient to allow the engine to develop up to 900 shp. This is considered adequate single-engine performance as it represents 72 percent of the main transmission power limit at the recommended topping rotor rpm.

Table VII

SINGLE-ENGINE POWER AVAILABLE LOSS SUMMARY

Percent Loss at Sea Level
from Standard Day to 35 Degrees C

Data Basis	Loss (pct)
T400 Engine Specification 712B	-12.7
NAVY (NAPTC) Test	-16
AFFTC - Engine topped on N_g	-13
AFFTC - Engine topped on ITT	-17

Power Available Determination

Correlating the topping power data was a difficult task. Category II testing was bound by the Technical Order (T.O.) information regarding topping power checks which was received concurrently with field distribution. Because the early T.O. data were optimistic, many adjustments were made to the fuel control governors (N_f and N_g), the ITT bias system, and the torque control system to obtain T.O. topping power. These adjustments probably contributed to some of the scatter that can be observed in the engine plots in appendix I. The cockpit procedures used to top the engines were the same as those described in operational supplement T.O. 1H-1(U)N-6CF-1S-5 (reference 6), and this procedure is considered satisfactory.

While determining power available, several important facts were discovered which warrant discussion. To determine power available, an accurate relationship of indicated torque to output torque is essential. This relationship involves two systems: the hydromechanical torquemeter relationship of pounds per square inch (psi) to output shaft torque (Q), and the electrical transmitter system which changes psi to indicated torque. The Category II engines were calibrated so that the psi-versus-Q relationship was accurately known. Among the four power sections at high power settings (near topping), the Q varied 2.5 percent (figures 4, 5, 7, and 8, appendix II). This means that if the system design relationship (Q versus psi) was the average of these values, an initial $\pm$1.25-percent error exists in the T.O. charts.

Figure 166, appendix I, shows pilot's panel indicated N_g and ITT values compared to those recorded on the special instrumentation. If the special instrumentation indications are assumed to be the more cor-

rect, a 1.5 percent low reading in N_g and a 10 degrees C low reading in ITT were realized. These deviation values (N_g, ITT, and Q) correspond with the accuracies reported in ASD/SDQH letter 7-85, T400 Engine Meeting, 19 July 1971.

Low ITT and N_g can mean a substantial power loss at topping power. Considering power section S/N 66127, for example, which limited on N_g, if 100-percent N_g was attained at sea level standard day, then 67.5-percent torque would be realized. If the indicator were reading 100 percent and the engine were actually at 98.5-percent N_g, 62.5-percent torque would be realized. This 5-percent difference in torque would probably mean rejection of a good power section. If an engine was limiting prematurely on ITT, that is, 800 degrees C instead of 810 degrees C, approximately 2.5 percent less torque would be produced. Since the indicating systems can drastically affect the indicated power output, frequent calibrations of the torque, ITT, and N_g systems should be made so that an indicating problem will not cause the replacement of an otherwise satisfactory engine. **(R 5)**

As a result of contractor information supplied at the December 1971 T400 engine meeting, topping power checks were made wherein topping power was maintained for 5 minutes. The contractor maintained that the compressor blades would elongate after approximately 4 minutes, making the engine more efficient, and more torque would be realized for given ITT and N_g conditions.

Stabilization occurred after approximately 4 minutes as is shown in figures 164 and 165, appendix I, but the behavior of the stabilization was again a function of which parameter the engine topped on. When the engine topped on N_g, torque remained constant, but ITT typically dropped 25 degrees C. When the engine topped on ITT, N_g increased approximately 1 percent and torque increased 2.5 percent. Due to a possible reduction in engine life from prolonged operation at high power settings, the time/topping check should be accomplished only if the engine output is marginally acceptable. **(R 6)**

For all topping checks it is very important to keep constant the maximum possible number of parameters which influence power. As a result, topping should be done during hover or on the ground if possible. If this is not possible the check should be made in level flight. Climb checks should be made only if checks using the other flight modes are not possible. **(R 7)**

CONCLUSIONS AND RECOMMENDATIONS

The UH-1N Category II performance tests showed that the test aircraft performance generally exceeded that estimated in the Flight Manual. A notable exception to this was level flight at low gross weights and/or altitude where the tests showed lower performance than the Flight Manual. Single-engine performance was good and enhanced the operational capability of the helicopter.

The data obtained in this test program are characteristic of the UH-1N series helicopter.

1. The data obtained in this test program should be incorporated into the Flight Manual.

The Flight Manual, Section II, does not include takeoff techniques utilizing rotor speed bleed.

2. The discussion on page 13 of this report should be included in the Flight Manual.

Single-engine operation may increase level flight maximum range more than 25 percent and loiter time 25 percent.

3. When the operational situation requires the maximum possible range and/or loiter time, single-engine operation should be used (page 18).

The Flight Manual contains no discussion on the techniques following an engine failure near the AVOID or CAUTION areas of the height-velocity curve.

4. The discussions on pages 40 and 41 of this report should be included in the Flight Manual.

Considerable difficulty was experienced in properly adjusting the engines to obtain topping power. Errors in the engine indicating systems can drastically affect the indicated power output and may cause the unnecessary replacement of a satisfactory engine.

5. The engine torque, ITT, and N_g indicating systems should be calibrated frequently (page 45).

Time/topping power checks require prolonged operation of the engine at high or maximum power settings which could result in a reduction in engine life.

6. The time/topping power check should be accomplished only if the engine output is marginally acceptable (page 45).

Engine topping power is governed by ITT, N_g or fuel flow, depending on atmospheric conditions and the flight modes which affect compressor inlet conditions. To obtain proper topping power indications, the conditions which influence power should be held constant.

7. Engine topping power checks should be made during hover or on the ground if possible. If this is not possible the check should be made in level flight. Climb checks should be made only if checks using the other flight modes are not possible (page 45).

Large airspeed position errors exist in low speed level flight, climb, and autorotational descent.

8. Means of reducing airspeed indicating errors should be investigated (page 2).

REFERENCES

1. Instrument Systems, Pitot-Static Tube Operated, Installation of, Military Specification, MIL-I-5072A, 14 April 1961.

2. Flight Manual, UH-1N Helicopter, T.O. 1H-1(U)N-1, March 1971.

3. Helicopter Flying and Ground Handling Qualities; General Requirements for, Military Specification, MIL-H-8501A, 7 September 1961.

4. Somsel, John R., and Thomas, David W., Captain USAF, Determination of the Effects of Rotor Blade Compressibility on the Performance of the CH-47A Helicopter, FTC-TR-66-46, Air Force Flight Test Center, Edwards AFB, California, March 1967.

5. Russell, Edward B., Major USAF, and Brandt, Jerome C., UH-1N Category II Propulsion System Evaluation, FTC-TR-71-39, Air Force Flight Test Center, Edwards AFB, California, August 1971.

6. Functional Check Flight Procedures, USAF Series UH-1N Helicopter, T.O. 1H-1(U)N-6CF-1, 1 March 1971, as supplemented.

BIBLIOGRAPHY

Springer, Robert H., Nasal, Timothy P., Grawi, Franklin L., Captain USAF, Flanigen, Edwin G., Lieutenant Colonel USAF, Angle, Theodore G., Major USAF, Air Force Preliminary Evaluation of the UH-1N Helicopter, AFFTC Report FTC-TR-70-22, August 1970.

Detail Specification for Twin Engine Helicopter, Bell Helicopter Report No. 212-947-010, 16 October 1969.

Model Specification, Engine, Aircraft, Turboshaft, Twin Power Section, YT400-CP-400 and T400-CP-400, United Aircraft of Canada Limited, UACL Spec. No. 712, 16 September 1968.

Berger, Donald, Lieutenant Colonel USAF, and Nasal, Timothy P., UH-1N Category II Flying Qualities Evaluation, FTC-TR-71-50, Air Force Flight Test Center, Edwards AFB, California, November 1971.

Berger, Donald, Lieutenant Colonel USAF, and Nasal, Timothy P., UH-1N Category II Flying Qualities Evaluation, FTC-SD-71-50, Air Force Flight Test Center, Edwards AFB, California, January 1972.

Angle, Theodore E., Major USAF, and Ford, James A., UH-1N Category II Tropical Weather Tests, FTC-TR-71-1, Air Force Flight Test Center, Edwards AFB, California, February 1971.

Nelsen, Gary L., Major USAF, and Ford, James A., UH-1N Category II Desert Tests, FTC-TR-71-48, Air Force Flight Test Center, Edwards AFB, California, March 1972.

Nelsen, Gary L., Major USAF, and Ford, James A., UH-1N Category II Arctic Tests, FTC-TR-72-10, Air Force Flight Test Center, Edwards AFB, California, March 1972.

Nelsen, Gary L., Major USAF, and Ford, James A., UH-1N Category II Climatic Laboratory Tests, FTC-TR-71-36, Air Force Flight Test Center, Edwards AFB, California, August 1971.

Russell, Edward B., Major USAF, and Helmick, Hugh M., Captain USAF, UH-1N Category II Airframe and Subsystems Evaluation, FTC-TR-71-37, Air Force Flight Test Center, Edwards AFB, California, November 1971.

Unclassified
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Air Force Flight Test Center Edwards AFB, California	2a. REPORT SECURITY CLASSIFICATION Unclassified 2b. GROUP N/A

3. REPORT TITLE
6 Category II Performance Test of the UH-1N Helicopter. Volume I.

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)
9 Final rept. 17 Nov 70-16 Feb 72,

5. AUTHOR(S) (First name, middle initial, last name)
10 Robert H. Springer
Donald Berger Lieutenant Colonel, USAF

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
11 May 1972 12 63p.	~~48~~	6

8a. CONTRACT OR GRANT NO. b. PROJECT NO. 16 AFFTC Directive 69-49B c. Program Structure 443N d.	9a. ORIGINATOR'S REPORT NUMBER(S) 14 AFFTC-TR-72-17-Vol-1 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A

10. DISTRIBUTION STATEMENT Distribution limited to U.S. Government agencies only (Test and Evaluation), February 1972. Other requests for this document must be referred to ASD (SDQH), Wright-Patterson AFB, Ohio 45433.

11. SUPPLEMENTARY NOTES This is Volume I of II	12. SPONSORING MILITARY ACTIVITY 6510th Test Wing Edwards AFB, California

13. ABSTRACT This report presents the results of the UH-1N Category II performance tests conducted to obtain data for the Flight Manual. In general, hover, climb, level flight, and takeoff performance equalled or exceeded that estimated in the Flight Manual; the exception being level flight at low weight and/or altitude. Level flight tests with external armament resulted in a 5- to 10-percent reduction in range capability depending on loading. The UH-1N had excellent single-engine performance resulting in a relatively small AVOID area on the height-velocity curve. A single-engine go-around was possible at all conditions outside a well defined CAUTION area. Slope landing tests were made on slopes up to 17 degrees. The standard airspeed system would not register airspeeds below 15 to 20 knots, and there were position errors of up to 9 knots in level flight and 7 knots in climb. Discrepancies in the engine power indicating systems were found to be sufficient to possibly cause an unnecessary replacement of a satisfactory engine.

DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

012100

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
UH-1N helicopter performance tests hover climb level flight takeoff range slope landing						

UNCLASSIFIED
Security Classification